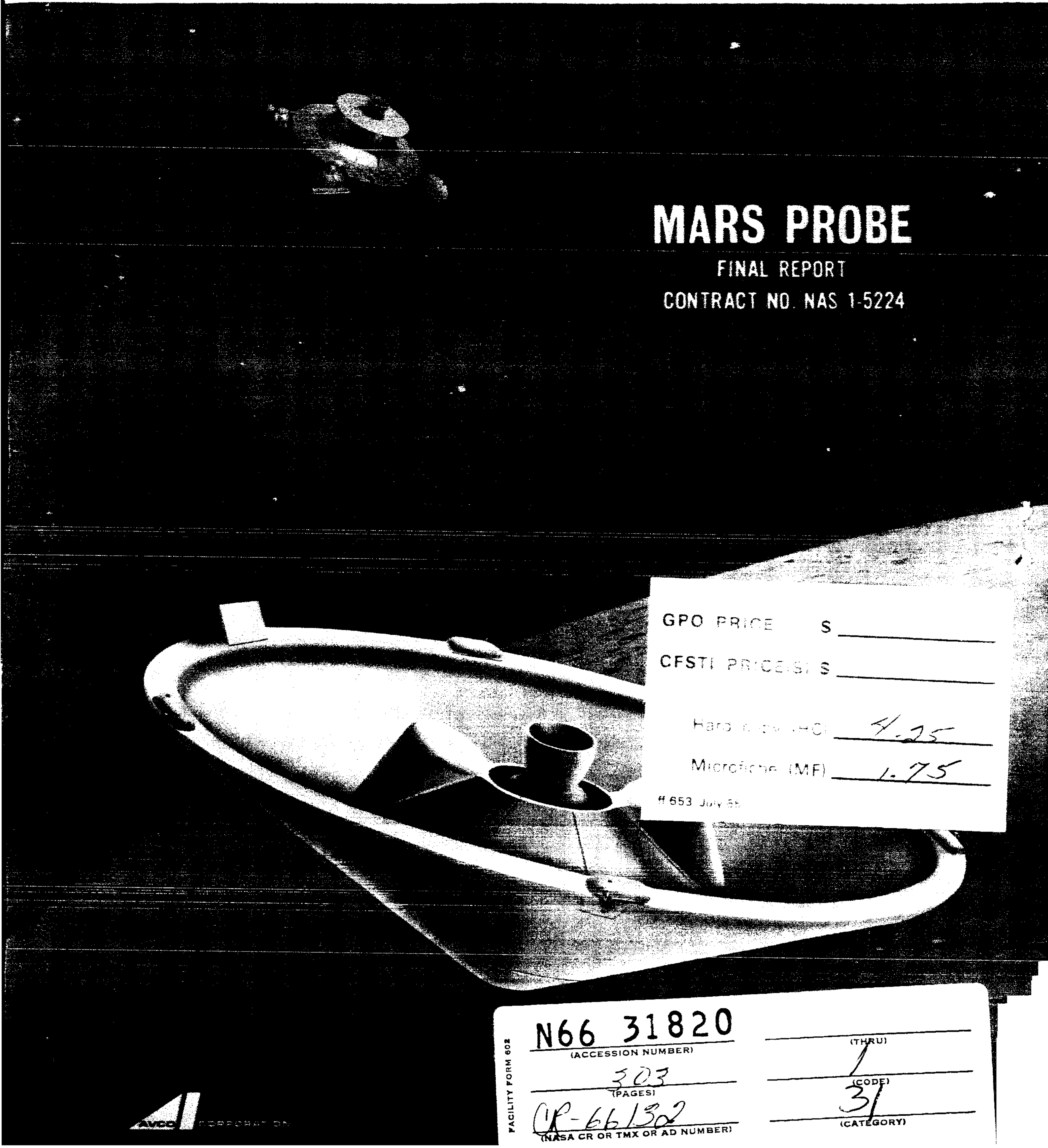
NASA CR-66132

VOLUME III PROBE, ENTRY FROM ORBIT

BOOK 2 **MISSION, SYSTEM and COMPONENT SPECIFICATIONS**



MARS PROBE

FINAL REPORT

CONTRACT NO. NAS 1-5224

AVCO CORPORATION

BOOK INDEX

NASA CR-66132

COMPARATIVE STUDIES OF CONCEPTUAL DESIGN AND QUALIFICATION PROCEDURES FOR A MARS PROBE/LANDER

FINAL REPORT

VOLUME III PROBE ENTRY FROM ORBIT
Book 2 MISSION SYSTEM AND SUBSYSTEM SPECIFICATIONS

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PREFACE

The results of Mars Probe/Lander studies, conducted over a 10-month period for Langley Research Center, NASA, are presented in detail in this report. Under the original contract work statement, studies were directed toward a direct entry mission concept, consistent with the use of the Saturn IB-Centaur Launch Vehicle, wherein the landing capsule is separated from the spacecraft on the interplanetary approach trajectory, some 10 to 12 days before planet encounter. The primary objectives of this mission were atmospheric sampling by the probe/lander during entry and terrain and atmosphere physical composition measurement for a period of about 1 day after landing.

Studies for this mission were predicated on the assumption that the atmosphere of Mars could be described as being within the range specified by, NASA Mars Model Atmospheres 1, 2, 3 and a Terminal Descent Atmosphere of the document NASA TM-D2525. These models describe the surface pressure as being between 10 and 40 mb. For this surface pressure range a payload of moderate size can be landed on the planet's surface if the entry angle is restricted to be less than about 45 degrees.

Midway during the course of the study, it was discovered by Mariner IV that the pressure at the surface of the planet is in the 4 to 10 mb range, a range much lower than previously thought to be the case. The results of the study were re-examined at this point. It was found that retention of the direct entry mission mode would require much shallower entry angles to achieve the same payloads previously attained at the higher entry angles of the higher surface pressure model atmospheres. The achievement of shallow entry angles (on the order of 20 degrees), in turn, required sophisticated capsule terminal guidance, and a sizeable capsule propulsion system to apply a velocity correction close to the planet, after the final terminal navigation measurements.

Faced with these facts, NASA/LRC decided that the direct entry from the approach trajectory mission mode should be compared with the entry from orbit mode under the assumption that the Saturn 5 Launch Vehicle would be available. Entry of the flight capsule from orbit allows the shallow angle entry (together with low entry velocity) necessary to permit higher values of M/C_DA, and hence entry weight in the attenuated atmosphere.

It was also decided by LRC to eliminate the landing portion of the mission in favor of a descent payload having greater data-gathering capacity, including television and penetrometers. In both the direct entry and the entry from orbit cases, ballistic atmospheric retardation was the only retardation means considered as apecifically required by the contract work statement.

Four months had elapsed at the time the study ground rules were changed. After this point the study continued for an additional five months, during which period a new design for the substantially changed conditions was evolved. For this design, qualification test programs for selected subsystems were studied. Sterilization studies were included in the program from the start and, based on the development of a fundamental approach to the sterilization problem, these efforts were expanded in the second half of the study.

The organization of this report reflects the circumstance that two essentially different mission modes were studied -- the first being the entry from the approach trajectory mission mode and the other being the entry from orbit mission mode -- from which two designs were evolved. The report organization is as follows:

Volume I, Summary, summarizes the entire study for both mission modes.

Volume II reports on the results of the first part of the study. This volume is titled Probe/Lander, Entry from the Approach Trajectory. It is divided into two books, Book 1 and Book 2. Book 1 is titled System Design and presents a discursive summary of the entry from the approach trajectory system as it had evolved up to the point where the mission mode was changed. Book 2, titled Mission and System Specifications, presents, in formal fashion, specifications for the system. It should be understood, however, that the study for this mission mode was not carried through to completion and many of the design selections are subject to further tradeoff analysis.

Volume III is composed of three books which summarize the results of the entry from orbit studies. Books 1 and 2 are organized in the same fashion as the books of Volume II, except that Book 2 of Volume III presents component specifications as well. Book 3 is titled Development Test Programs and presents, for selected subsystems, a discussion of technology status, test requirements and plans. This Book is intended to satisfy the study and reporting requirements concerning qualification studies, but the selected title is believed to describe more accurately the study emphasis desired by LRC.

Volume IV presents Sterilization results. This information is presented separately because of its potential utilization as a more fundamental reference document.

Volume V presents, in six separate books, Subsystem and Technical Analyses. In order (from Book 1 to Book 6) they are:

Trajectory Analysis
Aeromechanics and Thermal Control
Telecommunications, Radar Systems and Power
Instrumentation
Attitude Control and Propulsion
Mechanical Subsystems

Most of the books of Volume V are divided into separate discussions of the two mission modes. Table of Contents for each book clearly shows its organization.

CONTENTS

PART I FLIGHT CAPSULE MISSION AND SYSTEM SPECIFICATION

INTRODUCTION

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Cont'd)

CONTENTS (Concl'd)

ILLUSTRATIONS

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TABLES

ACKNOWLEDGEMENT

The conduct of the study and technical preparation of this report involved the participation and close coordination of many people, all of whose contributions were important to the end results.

It is impractical to single out each individual, but the major contributors to the study, reflected in the material presented in this Book, are as follows:

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INTRODUCTION

This specification provides the Capsule System general ground rules, guide lines and definitions of the general design philosophy, including the design criteria for the Flight Capsule (FC) and Operational Support Equipment (OSE), for the 1971 Mars opportunity wherein the Flight Capsule enters the Mars atmosphere after separation from a Planetary Vehicle (PV) which has attained an orbit around Mars for a predetermined time period. Primary functional areas are identified and functionally described.

With the establishment of this overall Project framework, the details of Capsule System operation within this framework followed. Detailed requirements data were formulated for each subsystem of the Capsule System and operational application of these requirements resulted in related subsystem design. The detailed subsystem descriptions are not necessarily the recommended design approach, but are intended to show all the factors which were considered and must be further analyzed and integrated for an effective total Capsule System concept.

An asterisk following any paragraph in this specification indicates the information was not available at the time the concept herein specified was synthesized or that the information was not critical to the synthesis.

PART I

FLIGHT CAPSULE MISSION

AND

SYSTEM SPECIFICATION

1.0 MISSION DESCRIPTION

The Flight Capsule mission objectives are defined in relation to the overall Mars exploratory mission, including a nominal flight profile.

1.1 FLIGHT CAPSULE MISSION OBJECTIVES

1.1.1 Primary

The primary mission objective of the Flight Capsule (FC) is to, 1) conduct experiments while in the Martian atmosphere to obtain atmospheric data and surface characteristics, and 2) transmit the results to Earth. The events required to accomplish this objective are ordered in the following manner:

1. Perform a successful launch and injection of the Planetary Vehicle (PV) (Flight Capsule and Flight Spacecraft (FS)) into a prescribed transfer trajectory.

2. Perform a successful injection of the Planetary Vehicle into an acceptable Martian orbit.

3. Perform a successful Flight Spacecraft - Flight Capsule separation at a prescribed time and location.

4. Deorbit and place the Flight Capsule on a preselected impact trajectory to Mars.

5. Enter the Martian atmosphere and obtain data on Martian atmosphere and surface characteristics from the Flight Capsule instrumentation and communicate these data to the Flight Spacecraft.

Additionally, the Flight Capsule mission will develop and provide experience in the use of the basic capability to enter the Martian atmosphere and impact the Martian surface; also the Flight Capsule will conduct observations that relate to critical design parameters for a future lander mission.

1.1.2 Secondary

A secondary objective is to provide experience with both the flight and ground systems required to deliver and operate the Flight Capsule payload.

1.1.3 Tertiary

A tertiary objective is to provide flight and ground equipment design compatible with subsequent missions.

1.1.4 Data Requirements

The primary mission objectives can be expanded in the area of data requirements and divided according to five major mission phases:

1. Preseparation
2. Separation
3. Separation to entry
4. Entry to chute deployment
5. Chute deployment to impact.

The engineering mission objectives include experiments to obtain the following data:

1. Preseparation

None.

2. Separation

None.

3. Separation to entry

Trapped radiation within Mars magnetosphere

4. Entry to parachute employment

Prime emphasis shall be placed on obtaining atmospheric data; redundant instrumentation shall be utilized as required. Data as a function of altitude shall be obtained and transmitted for the following:

a. Atmospheric density
b. Atmospheric pressure
c. Atmospheric temperature
d. Atmospheric composition
e. Trapped radiation within the atmosphere
f. Ionosphere electron density

5. Parachute employment to Impact

Prime emphasis shall be placed on obtaining atmospheric and Martian surface characteristics; redundant instrumentation shall be utilized as required. Data as a function of altitude shall be obtained and transmitted for items a. through g. of the following:

a. Chute deployment conditions

b. Atmospheric density

c. Atmospheric pressure

d. Atmospheric temperature

e. Atmospheric composition

f. Trapped radiation

g. Wind speed

h. Surface hardness and density

i. Surface roughness

j. Terrain characteristics

1.2 MISSION DEFINITION

As presently defined, the Capsule System mission shall include all the activities required to develop, integrate, test, launch, and operate one space vehicle consisting of two totally integrated Planetary Vehicles during the 1971 Mars launch opportunity. The mission is completed when all of the engineering and diagnostic data have been returned to Earth, reduced to acceptable form and disseminated to all cognizant organizations.

Air Force Eastern Test Range (AFETR) and the Kennedy Space Center (KSC) facilities at Cape Kennedy, Florida shall be utilized for prelaunch and launch activities. Prelaunch assembly, checkout and test shall be conducted in the Flight Capsule portion of the Flight Spacecraft facility. The Explosive Safe Facility (ESF) shall be furnished for all operations involving the Flight Capsule that are considered of a hazardous nature, such as terminal sterilization, propellant loading and hazardous component activation.

AFETR tracking and telemetry facilities shall be required during the launch through injection phase of the mission to accommodate the Space Vehicle instrumentation and assist in the Deep Space Network (DSN) acquisition.

The Capsule System is that portion of the overall project which accomplishes the planet atmospheric and surface experiments with an atmospheric Entry Vehicle and landed penetrometer experiment.

The integration and further definition of the Project systems to the Capsule System is discussed in paragraph 1.3.1.

1.3 MISSION ELEMENT DEFINITION

The successful accomplishment of the previously defined Capsule System mission objective requires the implementation of six major systems. Each system includes the operational hardware, software, and spares; related Operational Support Equipment (OSE); related development test models and facilities; all phases of design, development, fabrication and test; and all personnel assigned to activities required to support the prelaunch, launch and flight operation mission phases. These systems are as follows:

1. Launch Vehicle System
2. Spacecraft System
3. Capsule System
4. Mission Operations System
5. Tracking and Data System
6. Launch Operations System

1.3.1 Project Definition

The relation of the Capsule System to the rest of the Project is shown in Figure 1. The detailed definition of all of the Flight Capsule subsystems is presented in the appropriate end item functional specifications presented in Section 6.0. A general definition of the primary project elements follows:

1.3.1.1 Capsule System

The Capsule System includes the Flight Capsule (FC), as the flight hardware; plus flight hardware spare parts (or spare Flight Capsules depending on the time of spares replacement), development and sterilization assay models, qualification models, control documentation and associated software, Operational Support Equipment, and the management and engineering teams. The Flight Capsule consists of the required entry and descent equipment, engineering instrumentation payload, sterilization canister, Flight Capsule to Flight Spacecraft adapter and all other auxiliary subsystems hardware.

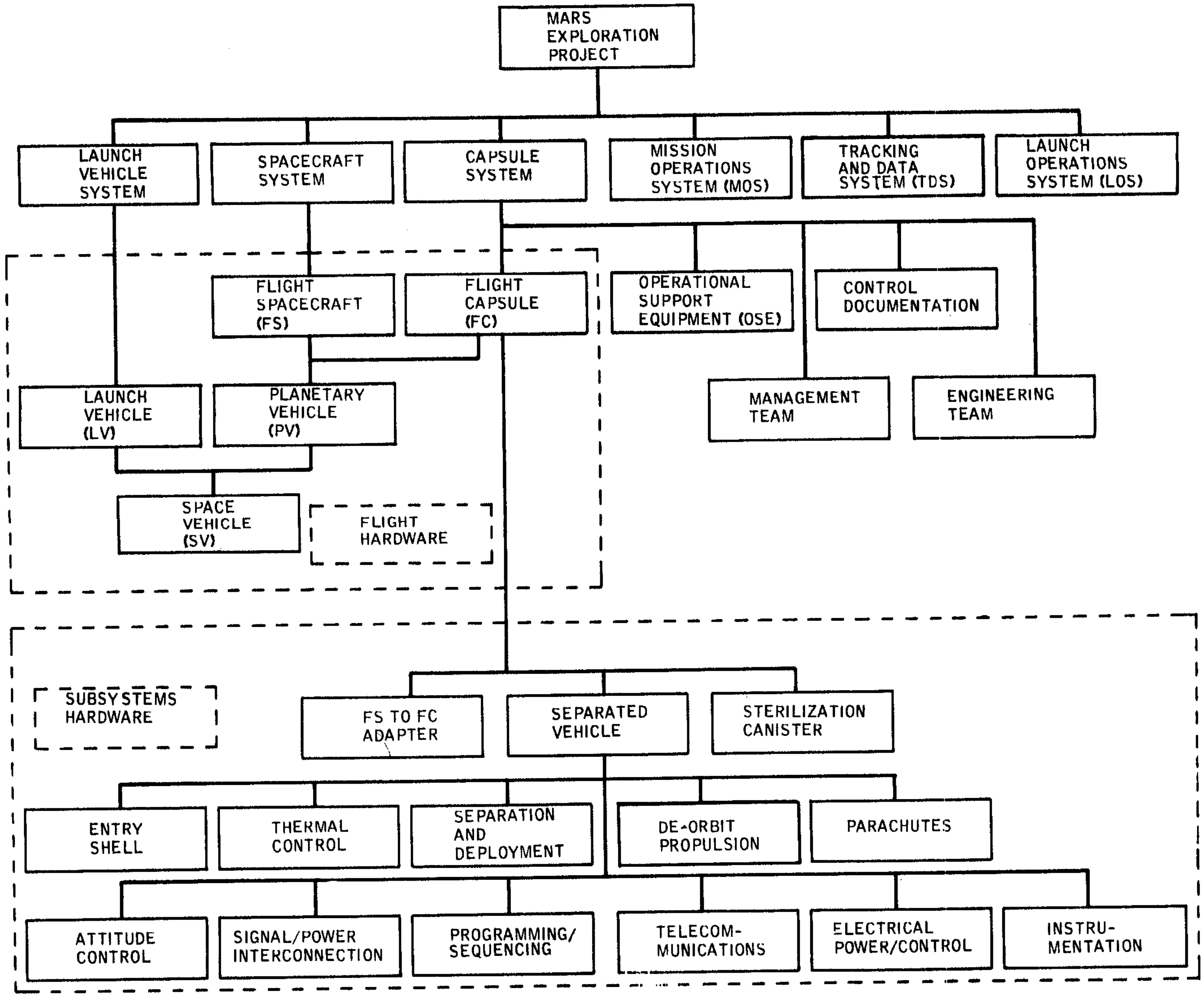

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Figure 1 CAPSULE SYSTEM FUNCTIONAL DIAGRAM AND TERMINOLOGY

1.3.1.2 Spacecraft System

The Spacecraft System includes the Flight Spacecraft (FS), as its flight hardware; plus flight hardware spare parts, development models, qualification models, associated Operational Support Equipment (hardware and software), and planetary vehicle adapter, the Spacecraft System test complex, all test facilities required for Flight Spacecraft and Planetary Vehicle testing, all facilities at KSC required to assemble the Flight Spacecraft and Planetary Vehicle, the Flight Spacecraft prelaunch and launch checkout equipment at the launch complex and all personnel required to fulfill the above functions. The Flight Spacecraft consist of a Spacecraft Bus, a Spacecraft science payload and a spacecraft propulsion subsystem.

1.3.1.3 Planetary Vehicle (PV)

The Planetary Vehicle is defined as the composite Flight Spacecraft and Flight Capsule integrally attached and operated up to separation in the vicinity of the selected planet. The Flight Spacecraft to Flight Capsule adapter includes the inflight separation joint between the Flight Capsule and Flight Spacecraft, the load path for mounting the Flight Capsule to the Flight Spacecraft and a mounting bulkhead for the umbilical connections between the Flight Capsule and Flight Spacecraft.

1.3.1.4 Launch Vehicle System

The Launch Vehicle System includes the three stages of the Saturn V with its guidance subsystems and the ascent fairing which shrouds the two Planetary Vehicles to make up the Launch Vehicle (LV) as the flight hardware (shown in Figure 2); plus the supporting ground equipment, software and associated manpower.

1.3.1.5 Space Vehicle (SV)

The Space Vehicle is the combined Launch Vehicle and Planetary Vehicles which physically leave the launch pad in conduct of the mission.

1.3.1.6 Mission Operations System (MOS)

The MOS includes that portion of the Project which plans, directs, controls, and executes (with support provided by the Deep Space Network) the space flight operation after injection of the Planetary Vehicle on its trajectory, the Mission-Dependent Equipment (MDE) required at the Deep Space Network, and the operational teams,

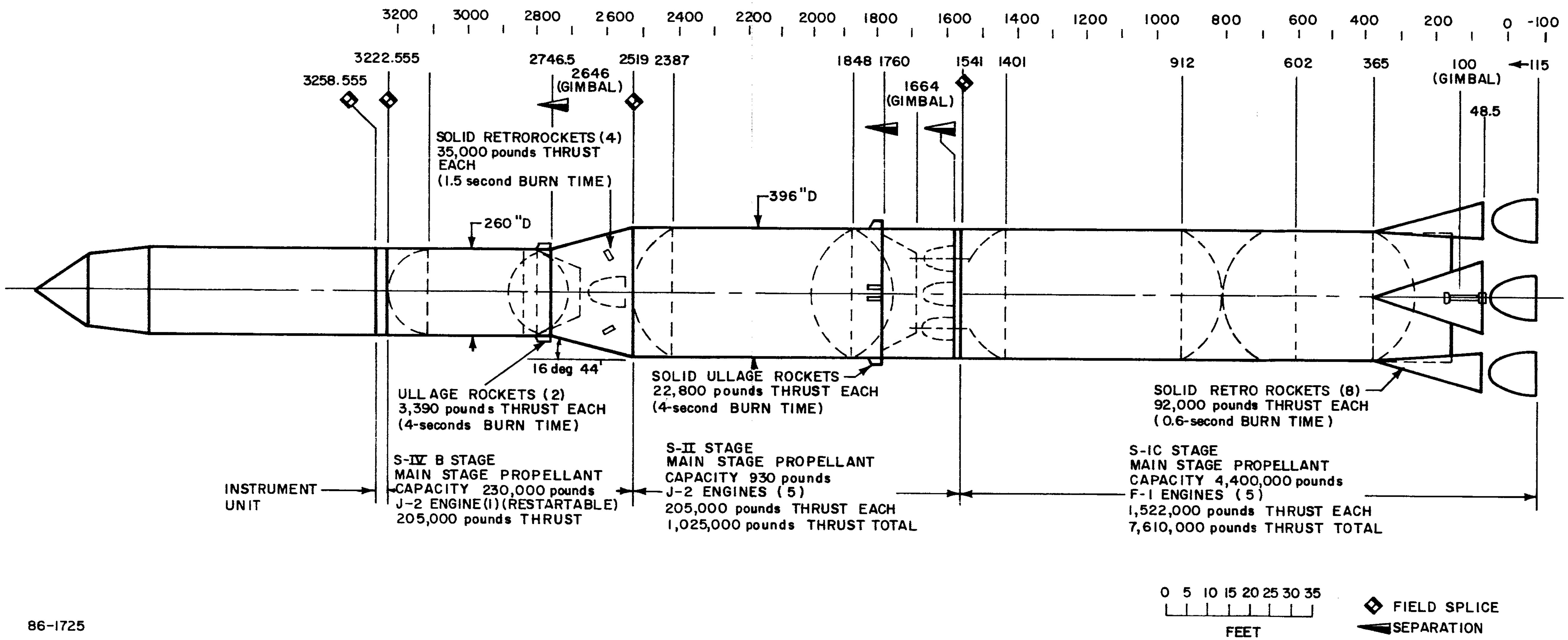

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Figure 2 LAUNCH VEHICLE CONFIGURATION

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1.3.1.7 Tracking and Data Systems (TDS)

The TDS includes the Deep Space Network (DSN) and all other NASA and Department of Defense (DOD) tracking and data acquisition stations, ships and aircraft assigned to support the mission, all NASA communications (NASCOM) and other circuits assigned to handle mission data and commands; and all other NASA and DOD personnel, facilities and equipment assigned to handle mission data, and command and support mission operations.

1. Deep Space Network (DSN) -- The DSN is comprised of the Deep Space Instrumentation Facility (DSIF), the Space Flight Operations Facility (SFOF), the Ground Communications System (GCS) connecting the two facilities, and the personnel who regularly operate these facilities. These facilities are defined in the following documents:

JPL TM No. 33-83, Revision 1

Deep Space Network - Space Flight Operations Facility Capability

Deep Space Network Ground Communications System Development Plan.

1.3.1.8 Launch Operations System (LOS)

The LOS includes those elements of the Project responsible for planning and executing the preflight and launch-to-injection phases of the mission.

1.3.1.9 Operational Support Equipment (OSE)

The OSE includes the equipment and facilities required for the assembly, servicing, checkout, sterilization, and testing of the subsystems of the Flight Capsule; described as follows:

1. Assembly, Handling and Shipping Equipment (AHSE) -- This equipment includes all lifting, holding, and positioning fixtures and other items required in the assembly and test of the Flight Capsule and its OSE, and for moving the Flight Capsule from place to place.

2. System Test Complex (STC) -- The STC is the basic set of test equipment used in tests and checkout to verify the adequacy of the Capsule System design, fabrication, assembly, and flight readiness. It shall be employed to monitor and record the performance of the total Capsule System and of its subsystems and subassemblies, while providing power, command, external stimuli, and simulation of the associated systems.

3. Launch Complex Equipment (LCE) -- The LCE shall be used to power and command the Flight Capsule and to monitor and record its functions (both by hardline and by telemetry) during prelaunch checkout while the Planetary Vehicle is installed in the Space Vehicle configuration on the launch pad. It includes all equipment on the launch pad and in the blockhouse. It shall also be used to support operations in the Explosive Safe Facility (ESF).

4. Mission Dependent Equipment (MDE) -- The MDE includes all items required at the site of the DSN to meet the functional requirements of a particular project which are not required on any other project. Items may be categorized as either software or hardware.

5. Maintenance Support Equipment (MSE) -- The MSE is the equipment required to support the Flight Capsule and its OSE such that the Capsule System maintains its capability of performing its mission.

6. Factory Support Equipment (FSE) -- The FSE is the equipment required to fabricate, assemble, and checkout the Flight Capsule and its support equipment. These equipments are limited to in-plant operations such as tooling and checkout gear.

7. Facilities -- The buildings that house the engineering and administrative personnel, test areas and chambers, manufacturing, assembly, training, and storage.

1.3.1.10 Control Documentation

Control documentation is the software required to define, integrate and test the Capsule System with the rest of the Project.

Typical documents in this category include, but are not limited to the following:

Model Specification

Electrical Interface Drawing

Mechanical Interface Drawing

Interference Control Specification

Interconnecting Cabling Diagram

Signal and Power Flow Diagram

Schematic Diagram

Storage and Handling Procedures

Integrated Test Plans and Procedures

1.3.2 Capsule System Major Interfaces

The major interfaces between the Capsule System and the other systems are as follows:

1.3.2.1 Capsule System - Spacecraft System

The Capsule System and Spacecraft System have physical, signal, power, RF and temperature control interfaces between the Flight Capsule and Flight Spacecraft portions of the overall system. The Flight Capsule shall be mounted on the forward end of the Flight Spacecraft within the envelope allowances as shown in Figure 3. The physical interface between the Flight Capsule and Flight Spacecraft shall be a field joint including the electrical connectors for interconnecting cables. The inflight separation system is part of the Capsule System and is forward (on the Flight Capsule side) of the field joint. All loads transmitted from the Flight Capsule to the Flight Spacecraft shall be across the field joint; no other structure support shall be provided for the Flight Capsule. All of the power and temperature control requirements of the Flight Capsule shall be supplied by the Flight Spacecraft from launch until Flight Capsule - Flight Spacecraft separation. The Flight Capsule - Flight Spacecraft separation sequence shall be initiated by the Flight Spacecraft. The Flight Spacecraft shall transmit all of the Flight Capsule telemetry data to Earth. Prior to Flight Capsule - Flight Spacecraft separation data shall be transmitted from the Flight Capsule to the Flight Spacecraft via hardline. After separation, the telemetry data shall be transmitted to the Flight Spacecraft via a radio relay link.

1.3.2.2 Capsule System - Operational Support Equipment

The Capsule System has both physical and functional interfaces between Operation Support Equipment in the System Test Complex and Launch Complex Equipment. The physical interface shall include the interconnecting cables.

1.3.2.3 Capsule System - Mission Operational System

The Capsule System has a functional interface with the Mission Operational System for the Capsule System data flow.

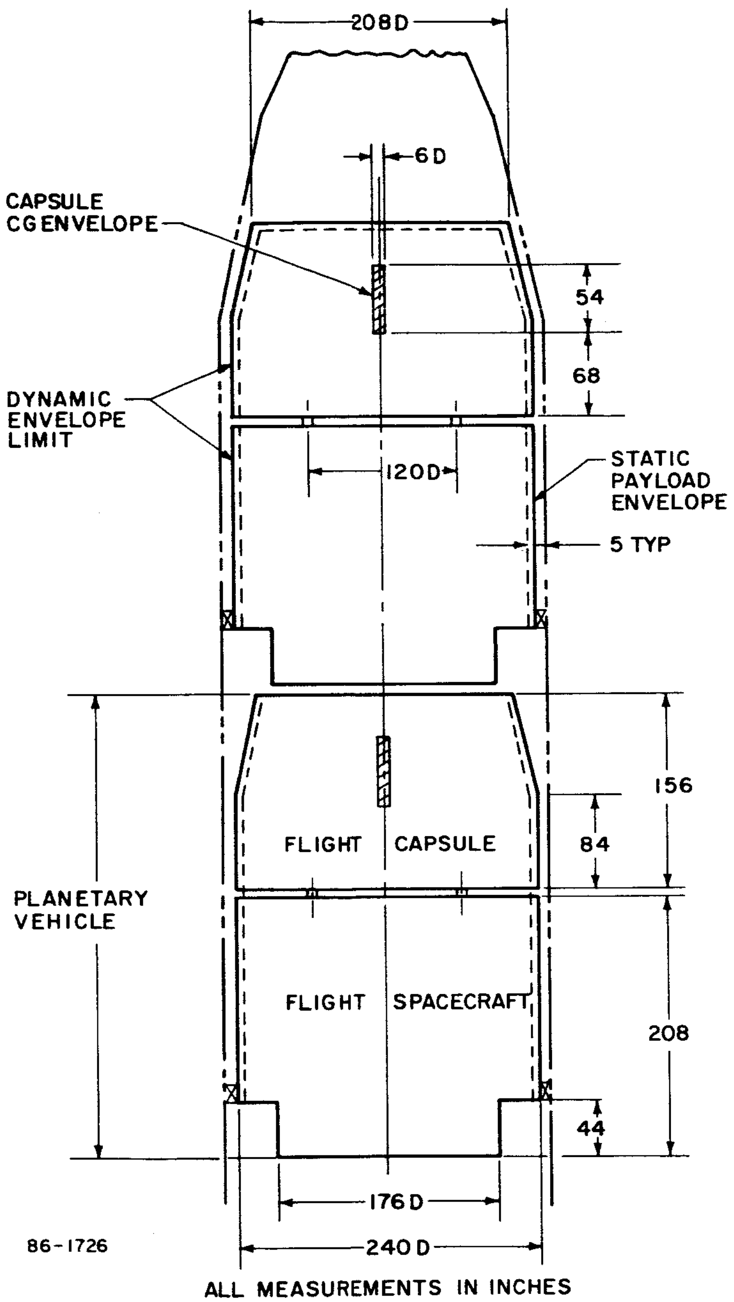

Figure 3 PLANETARY VEHICLE ARRANGEMENT AND DYNAMICS ENVELOPES

1.4 MISSION PROFILE

As presently defined, all mission oriented operational activities begin with mission acceptance test and review, and subsequent shipment of the flight hardware to Kennedy Space Center. Flight Capsule mission operations will terminate when all Flight Capsule acquired data has been received on Earth, reduced to acceptable form, and disseminated to all cognizant organizations. The events in a nominal mission profile are defined as follows:

1. Prelaunch	Space Vehicle final assembly and integration, Flight Capsule sterilization verification, systems test and all activities leading to commitment to launch.
2. Launch and transfer orbit injection	Space Vehicle countdown, launch insertion into parking orbit, injection into planetary transfer orbit and separation of the two Planetary Vehicles.
3. Interplanetary cruise	All events and sequences required to assure the proper interplanetary trajectory including reference acquisition and trajectory correction maneuvers.
4. Flight Capsule sterilization canister separation	All events and sequences required to separate the sterilization canister lid from the Planetary Vehicle.
5. Mars orbit insertion and preseparation operations	All events and sequences required to insert the Planetary Vehicle into a Mars orbit including orbital trim corrections as required.
6. Flight Capsule - Flight Spacecraft separation	All events and sequences required to separate the Flight Capsule from the Flight Spacecraft including the return of the Flight Spacecraft to orbital operational status.
7. Separated Vehicle de-orbit maneuver	All events and sequences required to place the Separated Vehicle on a preselected Martian impact trajectory.
8. Separated Vehicle orbital descent	All events and sequences required during the time period of Separated Vehicle de-orbit to Entry Vehicle entry.

9. Entry Vehicle entry	All events and sequence required during the time period that the Entry Vehicle reaches an altitude of 800,000 feet above the Martian surface until parachute deployment for terminal descent.
10. Suspended Capsule terminal descent	All events and sequences required during the time period from parachute deployment until impact on the Martian surface.

2.0 MISSION CONSTRAINTS AND COMPETING CHARACTERISTICS

2.1 MISSION CONSTRAINTS

The following items are the major constraints placed on the Capsule System by the mission and shall be considered fixed.

2.1.1 Mission Schedule

Since the mission objectives involve the 1971 Mars opportunity all designs, techniques, and components must be compatable with the schedule and all subsequent milestones based on that opportunity date.

2.1.2 Planetary Quarantine

The probability of Mars contamination for that portion of the mission related to a single Flight Capsule, shall be less than 10^{-4}.

2.1.3 Number of Launches

One Space Vehicle shall be launched for the 1971 Mars opportunity. The Space Vehicle shall be made up of two Planetary Vehicles plus one Launch Vehicle.

2.1.4 Launch Site

The Space Vehicle shall be launched from Launch Complex 39 at the Kennedy Space Center.

2.1.5 Launch Azimuth

The launch azimuths to be utilized for the mission shall be between 60° east of north and 115° east of north.

2.1.6 Launch Period

A launch period of 30 to 60 days will be provided, allowing for a minimum daily firing launch window of two hours.

2.2 COMPETING CHARACTERISTICS

The following list represents the priority sequence of the mission characteristics, in decreasing order of importance:

1. Probability of success

2. Performance of Mission Objectives

3. Cost

4. Contribution to future missions

5. Additional 1971 mission capability

The mission characteristics are in turn interpreted into Flight Capsule system characteristics, which are presented in the following list and shall be used to govern the priority of competing technical requirements:

1. Planetary sterility

2. Telecommunication functions and operations

3. Flight Capsule temperature control

4. Flight Capsule electrical power

5. Flight Capsule - Flight Spacecraft separation

6. Separated Vehicle de-orbit into proper Mars entry trajectory

7. Operation of instrumentation while in Martian atmosphere

8. Entry and descent of Entry Vehicle

9. Application of design philosophy, and flight and ground hardware to future missions.

3.0 MISSION REQUIREMENTS

3.1 GENERAL

The general requirements of this section apply to all systems required to perform the mission with particular emphasis on the mission requirements of the Capsule System.

3.1.1 Commitment to Launch

The Capsule System shall utilize instrumentation necessary to perform monitoring during countdown to detect malfunctions and nonstandard system operation. In the event of nonstandard performance, the system shall be capable of evaluating its effect on mission performance and shall affirm commitment to launch or request hold for maintenance as prescribed by the mission launch and hold criteria in effect at the time of launch.

3.1.2 Mission Success

All Capsule System equipment shall utilize design, manufacturing, test, operational techniques and procedures designed to maximize the probability of mission success and, in the event of failures or malfunctions other than catastrophic, produce partial success. To obtain this objective, consideration shall be given but not limited to the following items:

1. Failure mode and effect analyses and partial mission success in the event of failures.

2. Adequate design margins.

3. Block and functional redundancy techniques to minimize the effects of failure and increase overall system reliability.

4. Identification and elimination of unreliable items.

The philosophy of design shall emphasize simplicity and conservatism and shall include a complete and integrated test program for components, subsystems, and system to assure the highest degree of mission success. In addition, simplification of interfaces among the Capsule System individual subsystems, as well as among the Flight Capsule, the Flight Spacecraft, the Launch Vehicle, the Mission Operation System, and the Trajectory and Data System, shall be stressed to provide a simplified design, flight operations, ground test, checkout operation and thus maximize mission success. Wherever possible, previous experience gained on other programs related to hardware and development techniques should be applied.

For purposes of evaluating mission success, the Flight Capsule mission is divided into three major phases:

1. Launch and Transit Phase

2. Martian Orbit Injection Phase

3. Mars Entry Phase

Listed below are the major functions associated with each phase along with the cumulative probability-of-success goal.

3.1.2.1 Launch and Transit Phase

1. Successfully launch and inject one Space Vehicle into prescribed interplanetary transfer orbits. Probability of success goal= __*__.

2. Successfully separate two Planetary Vehicles from the Space Vehicle and successfully acquire attitude reference and cruise with at least one of the Planetary Vehicles including midcourse corrections as required. Probability of success goal = __*__.

3.1.2.2 Martian Orbit Injection Phase

1. Successfully inject at least one Planetary Vehicle into an acceptable Martian orbit, including orbital trim correction as required. Probability of success goal = __*__.

3.1.2.3 Mars Entry Phase

1. Successfully perform at least one Flight Capsule - Flight Spacecraft separation at a required time and location. Probability of success goal = __*__.

2. Successfully de-orbit at least one Flight Capsule and inject into a predetermined entry trajectory. Probability of success goal = __*__.

3. Successfully perform an entry into the Martian atmosphere and acquire and transmit data to Earth via relay communication to the Flight Spacecraft prior to Suspended Vehicle impact on the Martian surface. Probability of success goal = __*__.

3.1.3 Trajectories

3.1.3.1 Transfer Trajectories

The Planetary Vehicle shall be placed on a Type 1 transfer trajectory. A maximum value of C_3 is assumed to be 25 km^2/sec^2 and the maximum value of C_4 is assumed to be 20.25 km^2/sec^2. The absolute value of the Declination of the Launch Asymptote (DLA) shall not be less than 5 degrees and the inclination of the heliocentric transfer plane to the ecliptic shall not be less than 0.1 degrees.

3.1.3.2 Arrival Date Separation

There shall be at least 10 days separation in the arrival dates at Mars between the two Planetary Vehicles.

3.1.3.3 Injection and De-orbit over Goldstone

The Planetary Vehicle orbit injection and Flight Capsule de-orbit shall occur in view of the DSIF station at Goldstone, California.

3.1.3.4 Landing Site

Acceptable landing sites are Syrtis Major (80° east longitude; 10° north latitude) and Mare Sirenum (185° east longitude; 20° south latitude).

3.1.3.5 Orbit Selection

The orbit selected shall provide favorable conditions for the orbital operations of the Flight Spacecraft as well as providing the required conditions for Flight Capsule de-orbit. In addition, the orbit shall be such as to provide the required communication relay-link geometry and satisfy the landing site requirements.

3.1.3.6 Entry Conditions

The Entry Vehicle shall be designed to accommodate entry angles ranging from the skip-out limit of -8.5 to -15.1 degrees maximum, for the minimum entry velocity of 12,500 ft/sec at 800,000 feet entry altitude, and, the skip-out limit of -12.9 to -16.0 degrees maximum, for the maximum entry velocity of 15,200 ft/sec at 800,000 feet entry altitude.

3.1.4 Accuracy

3.1.4.1 Impact Point

The dispersion of the Suspended Capsule impact point shall not exceed 500 km (3σ) about the nominal aiming point with a design goal of 300 km (3σ).

3.1.4.2 Time from De-orbit to Entry

The dispersion in time from Separated Vehicle de-orbit to entry of the Entry Vehicle about the nominal time shall not exceed $\pm$ __*__ minutes (3σ), and shall be $\pm$ __*__ minutes (3σ) as a design goal.

3.1.5 Weight Allocation

The weight allocation for the Planetary Vehicle shall be as follows:

	Wt. -Lbs.
Spacecraft Bus (includes 400 pounds of science)	2,500
Spacecraft Propulsion	15,000
Flight Spacecraft	17,500
Flight Capsule (350 pounds remain with Flight Spacecraft	3,000
Planetary Vehicle	20,500
Planetary Vehicle Adapter	1,500
Total Weight, pounds.	22,000

3.1.6 Environmental Requirements

All Capsule Systems shall be required to operate and/or survive, as appropriate, in the natural and induced enviroments shown in Figure 4. Also, the Flight Capsule shall be required to operate and perform its required functions over the entire range of postulated Martian atmospheres, which properties are listed in Table I.

FLIGHT CAPSULE MISSION DESCRIPTION

Phase	Function	Duration
1. Manufacturing and Assembly	Manufacture, assembly and test of subsystems into the Flight Capsule for acceptance test; seal in canister, sterilize, checkout, and prepare for shipment.	up to 57 weeks
2. Storage and Transportation	OSE to be provided to maintain conventional conditions for the Flight Capsule/canister during all operations.	up to 13 weeks storage, up to 12 hours air transport.
3. Field Operations	Flight Capsule/canister integration with Flight Spacecraft. All operations in clean room.	up to 9 weeks.
4. Launch Pad Operations	Planetary Vehicles integration with Launch Vehicle. Clean room facilities on gantry prior to ascent fairing installation. Conditions maintained by ascent fairing after installation.	up to 4 weeks
5. Launch through Booster Separation	Planetary Vehicles are injected into interplanetary transfer orbit. Ascent fairing separated after atmospheric portion of boost.	up to 1 hour
6. Interplanetary Transit	Planetary Vehicle in unpowered trajectory except for attitude control maneuvers during acquisition and maintenance of attitude reference to Sun and Canopus, midcourse corrections with the Flight Spacecraft propulsion subsystem.	up to 384 days
7. Planetary Orbit	Flight Spacecraft propulsion subsystem main retro maneuver to acquire and maintain planetary orbit.	up to 30 days
8. Separation and Deorbit	Flight Capsule separates from the Planetary Vehicle; attitude control and retro propulsion maneuvers select entry corridor.	up to 1 hour
9. Atmospheric Entry	Entry Vehicle sensors acquire atmospheric data for subsequent playout.	up to 10 minutes
10. Atmospheric Descent	Additional scientific data is collected and played out with entry data while suspended from the parachute.	up to 15 minutes

Figure 4 SUMMARY OF FLIGHT CAPSULE MISSION PROFILE AND ENVIRONMENTAL CONDITIONS

ENVIRONMENTAL CONDITIONS

Generated	Electro-Magnetic Fields: Solar	Electro-Magnetic Fields: Particle	Electro-Magnetic Fields: Magnetic	Vibration	Shock	Sustained Acceleration	Acoustic Noise
Various forms of electro magnetic, fields may be generated by Flight Spacecraft, Flight Capsule or Launch Vehicle systems and/or OSE. Refer to MIL-I-266000 or RAD-E-59064	Negligible	Negligible	Earth field = 0.173 gauss. Vibration exciters = 80 oersteds. Tools = 100 oersteds.	Unpackaged (accel. RMS)(Freq cps) $\pm 3.5G_e$ 2 to 50 $\pm 1.5G_e$ 5 to 300 Packaged $\pm 1.3G_e$ 2 to 26 0.036 in. DA 26 to 52 $\pm 5.0G_e$ 52 to 300	Unpackaged components $100G_e$ for 6 ms. Unpack. maj. assemblies $10G_e$ for 11 ms applied along 3 axes or 1" flat drops and 4" pivot drops.	Earth gravity $G_e = 32.17$ ft/sec^2	Negligible
	Intensity varies sinusoidally from 0 to 0.140 watts/cm^2 in 24 hour periods spectrum percentage-8% ultraviolet, 41% visible, 51% infrared.		Earth field = 0.173 gauss	Unpackaged - no longer applicable; packaged - same as manufacture and assembly	Unpackaged assemblies & components: Not applicable. Packaged assembly with OSE MIL-P-7936		Related to transportation equipment and OSE attenuation.
					Unpackaged: not applicable. Packaged same as manufacturing and assembly		Negligible
	Max. 0.140 watt/cm^2 at 1 AU.		from 0.173 to less than 0.0013 gauss	Max. power spectral density (G_e^2/cps) = 0.2 from 200 to 500 cps	200 G_e peak saw tooth at 0.7 to 1.0 millisec rise time	maximum additional loads: axial = $4.7G_e$ Lateral = < 2 G_e	155 DB maximum from launch vehicle engines
	From 0.140 watts/cm^2 at 1.0 AU to 0.045 watts/cm^2 at 1.6 AU. Albedo factor: Earth 0.40, Mars 0.14. Light Pressure: 2x10^{-7}lb/ft^2 at 1.0 AU, 0.8x10^{-7}lb/ft^2 at 1.6 AU. Particle wavelength fractions of total energy vary through out entire spectrum.	Radiation dosages in space produced by atomic particles vary due to Earth "Van Allen" radiation belts, cosmic rays, solar flare activities and Mars radiation belts (if any). Max. energy and ionization field of protons, electrons & photons dependent on spacecraft trajectory versus time	Negligible	Negligible - to be verified by analysis of the selected FS midcourse correction propulsion subsystem versus PV dynamics.			
				To be determined from analysis of inter-relations of the FS propulsion subsystem effects on the PV dynamics		Axial: up to $2.9G_e$	To be determined
				To be determined from dynamics of FS and FC due to 1200 lbs. of separation forces and 2500 lbs. thrust from the FC propulsion subsystem.			
		Unknown probably negligible.	To be determined. assumed 0.5 earth.	To be determined from analysis of entry dynamics, Entry Vehicle configuration, deceleration device (parachute) mode of operations, atmosphere conditions, etc.			
	Some attenuation of short wavelengths in the atmosphere			negligible	negligible	Mars gravity, $G_m = 12.3$ ft/sec^2	None

Figure 4 (Cont'd)

23-1

2

EXPERIMENTAL CONDITIONS

Temperature (°F)	Pressure (mm Hg)	Relative Humidity	Sand and Dust	Precipitation	Corrosive Atmosphere	Fungus	Atmospheric Wind	Solar Wind	Meteoroids and Dust
60 to 80 during assembly and test, 275 for 24 hrs. during sterilization.	Sea Level ambient 760	40 to 50 percent	negligible	Not applicable	Ethylene oxide flush during sterilization cycles.	Non-fungus nutrients to be used for all equipments, sterilization further reduces any fungus probabilities internal to sterile canister, external assumed per paragraph 4. 8. 1 of MIL-E-5272.	Not applicable	Not applicable	Not applicable
-35 to 150 includes a 25-degree solar radiation rise.	760 to 87	0 to 100 percent	10^{-4} to 0. 3 mm diameter blown by 60-mph winds	Same as launch pad operations	Salt fog of 20 percent salt/80 percent water at specific gravity, 1. 126 to 1. 157 and a pH of 6. 5 to 7. 2 at 95°F 85 percent relative humidity		0 to 60 mph		
60 to 80	760	40 to 50 percent	negligible	Not applicable	None		Not applicable		
30 to 150 includes 25-degree solar radiation rise.	760	10 to 100 percent	10^{-4} to 0. 3 mm diameter blown by 60-mph winds	Rain: max. rate of 10 cm/hr blown by 60 mph wind. Ice: Up to a cm frozen on exposed surfaces. Snow: 1 to 3 mm diameter crystal at max. rate of 7. 5 cm/hr blown by 60 mph wind. Hail: 0. 5 to 5 mm diameter frozen pellets at max. rate of 10 cm/hr blown by 50 mph wind for 15 min. at above freezing temperature. Sleet: 0. 75 to 1. 5 mm dia. ice pellets at max. rate of 10 cm/hr blown by 60 mph wind for 30 min. at below freezing temperature.	Salt sea atmosphere of 5 percent salt/ 95 percent water		0 to 60 mph		
	760 to 10^{-c}	10 to 100% through none	16^{-4} to mm diameter blown by 60 winds to flexible		From salt atmosphere is none		Velocity versus altitude per ARDC Handbook		
Temperature of capsule system and subsystems to result from atmospheric ascent heating (max. aero-heat rate = 24. 2 watts/ft^2, solar radiation, internal equipment operations, space shadow effects, materials absorptivity/ emmisivity characteristics and related configuration. Analysis required for each critical area assuming no more than +300° at 1. 0 AU, to less than -300° in shadow at 1. 6 AU.	10^{-6} to 10^{-12}	None	negligible		None	None	Not applicable	At an AU (dynes/cm^2) 10^{-7} quiet sun 10^{-5} average 10^{-3} maj. activity	Near Earth model $N = (10^{-12})m^{-3/4}$ where N = impacts/ $meter^2$/sec of particles of mass m (grams) or greater; at an average velocity of 30 km/sec and a density of 8 gm/ cm^3 - 2% 3. 5 gm/ cm^3 - 10% 0. 44 gm/cm^3 - 88%. Near Mars model to be updated from Mariner 4 data.
To be determined	10^{-12} to 10 less than 10	unknown	unknown quantities of sand and dust blown by 100 ft/ sec winds.	H_2O and CO_2 crystals blown by 100 ft/sec winds.	improbable	unknown	To 200 ft/ sec gusts, for 10 seconds duration	Not applicable	Not applicable

Figure 4 (Concl'd)

TABLE I

SUMMARY OF STANDARD MODEL ATMOSPHERE PARAMETERS FOR MARS

Property	Symbol	Dimension	VM-3	VM-4	VM-7	VM-8
Surface pressure	P_o	mb lb/ft^2	10.0 20.9	10.0 20.9	5.0 10.4	5.0 10.4
Surface density	ρ_o	$(gm/cm^3)10^5$ $(slug/ft^3)10^5$	1.365 2.65	2.57 4.98	0.68 1.32	1.32 2.56
Surface temperature	T_o	°K °R	275 495	200 360	275 495	200 360
Stratospheric temperature	T_s	°K °R	200 360	100 180	200 360	100 180
Acceleration of gravity at surface	g	cm/sec^2 ft/sec^2	375 12.3	375 12.3	375 12.3	375 12.3
Composition:						
CO_2 (by mass)			28.2	70.0	28.2	100.0
CO_2 (by volume)			20.0	68.0	20.0	100.0
N_2 (by mass)			71.8	0.0	71.8	0.0
N_2 (by volume)			80.0	0.0	80.0	0.0
A (by mass)			0.0	30.0	0.0	0.0
A (by volume)			0.0	32.0	0.0	0.0
Molecular weight	M	mol^{-1}	31.2	42.7	31.2	44.0
Specific heat of mixture	C_p	cal/gm° C	0.230	0.153	0.230	0.166
Specific heat ratio	γ		1.38	1.43	1.38	1.37
Adiabatic lapse rate	Γ	°K/km °R/1000 ft	-3.88 -2.13	-5.85 -3.21	-3.88 2.13	-5.39 -2.96
Tropopause altitude	h_T	km kft	19.3 63.3	17.1 56.1	19.3 63.3	18.6 61.0
Inverse scale height (stratosphere)	β	km^{-1} $ft^{-1} \times 10^5$	0.0705 2.15	0.193 5.89	0.0705 2.15	0.199 6.07
Continuous surface wind speed	$\bar{v}$	ft/sec	155.5	155.5	220.0	220.0
Peak surface wind speed	v_{max}	ft/sec	390.0	390.0	556.0	556.0
Design vertical wind gradient	$\frac{d\bar{v}}{dh}$	ft/sec/1000 ft	2	2	2	2

3.1.7 Test Requirements

At a minimum, the following types of tests shall be required to be performed on the Capsule System.

3.1.7.1 Environmental Test

All Capsule System equipments shall be required to demonstrate their operability and/or survivability, as appropriate, under all environmental conditions as set forth in paragraph 3.1.6 above.

3.1.7.2 System Tests

System tests shall be conducted to demonstrate that each Flight Capsule is capable of performing its required function to meet the Mission Requirements. Each subsystem shall be required to demonstrate that the operational equipment is manufactured in accordance with the design as tested to assure functional compatibility.

3.1.7.3 Combined System Tests

Combined systems tests shall be required to demonstrate functional and physical compatibility with all interfacing systems. The objectives of these tests is to provide assurance of meeting mission requirements of the Capsule System.

3.2 CAPSULE SYSTEM REQUIREMENTS

3.2.1 General

The Capsule System shall be required to provide four fully qualified Flight Capsules for the Mars launch opportunity, the Operational Support Equipment (OSE) for integration and checkout of the Flight Capsule and the OSE for the prelaunch, launch and flight phases of the operational mission.

3.2.2 Functional

The Flight Capsule in the Separated Vehicle configuration, shall be capable of independent space flight from the point of separation from the Flight Spacecraft and de-orbit to entry into the Martian atmosphere. Operating subsystems shall be capable of surviving entry and descent to the Martian surface and will provide all power, thermal control, and communications requirements of the Flight Capsule engineering instrumentation. Specific functions to be performed by the Flight Capsule are as follows:

1. Provide the capability to perform an accurate de-orbit maneuver after the Flight Capsule has separated from the Flight Spacecraft and place the Entry Vehicle on a Martian surface impact trajectory.

2. Acquire data on Mars environment and surface characteristics.

3. Provide one-way communications capability to transmit Flight Capsule acquired data to the Flight Spacecraft for relay to Earth.

3.2.3 Performance

The Flight Capsule shall be capable of providing the following performance capabilities:

1. Capability shall be provided to operate and support up to 190 pounds of engineering instrumentation to measure the Martian environmental and surface characteristics during entry and descent.

2. Capability shall be provided to perform the separation, entry, and descent functions within the accuracies previously stated.

3. Capability shall be provided to impart a de-orbit incremental velocity to the Flight Capsule of 1400 ft/sec after the required de-orbit orientation has been obtained and to maintain the proper orientation throughout the incremental velocity application.

3.2.4 Sterilization

All flight hardware placed on a Martian impact trajectory shall be biologically clean per NASA planetary quarantine requirements. A sterilization canister shall be designed and built as part of the Flight Capsule to provide microbial barrier between the Flight Capsule and its surroundings. The Flight Capsule with the capsule engineering instrumentation payload will be placed inside the sterilization canister after final assembly, but prior to terminal sterilization, and sealed. The Flight Capsule will remain within the canister until such time as it is considered biologically safe to remove it prior to Flight Capsule - Flight Spacecraft separation. The sterilization canister lid shall be removed at such a time and manner so as not to compromise the Martian quarantine requirements or interfere with the operation of either the Flight Capsule or the the Flight Spacecraft.

3.2.5 Flight Spacecraft Requirements on Flight Capsule

3.2.5.1 The Flight Capsule center of gravity envelope shall be within those limits specified in paragraph 4.4. Figure 3 presents the geometric as well as the center of gravity constraints placed on the Flight Capsule by the Flight Spacecraft and the Launch Vehicle ascent fairing.

3.2.5.2 At separation, the Flight Capsule shall impart no moments or forces that will disturb the Flight Spacecraft so as to cause it to lose attitude reference.

3.2.5.3 The thermal control system of the Flight Capsule shall be designed to minimize the heat transfer to the Flight Spacecraft when in the Planetary Vehicle configuration.

3.2.6 Flight Capsule Requirements on Launch Vehicle

During the launch, parking-orbit and planetary transfer-orbit injection phases, the Launch Vehicle is required to functionally support the Planetary Vehicle. The Flight Capsule shall not require any unique capabilities over and above those needed for the Planetary Vehicle.

3.2.7 Flight Capsule Requirements on Flight Spacecraft

The Flight Spacecraft shall supply all power, thermal control, and communication requirements of the Flight Capsule during the time period after planetary transfer - orbit injection and prior to Flight Capsule - Flight Spacecraft separation. After separation the Flight Spacecraft will relay all Flight Capsule telemetry data to Earth.

3.2.8 Flight Capsule Requirements of Mission Operations System

The Mission Operations System is required to perform the following functions in direct support of the Flight Capsule mission:

1. Determine Flight Capsule separation location and time as a function of actual orbital parameters.

2. Determine Flight Capsule de-orbit maneuver.

3. Determine Flight Capsule time sequence of events to provide initiation commands for all entry events which require updating after launch.

3.2.9 Flight Capsule Requirements on the Tracking and Data System

Data acquired by the Flight Capsule will be received at the Space Flight Operations Facility (SFOF) for data processing and display. The communications, data processing and display facilities provided by the SFOF may require augmentation by Mission Dependent Equipment specifically implemented to the requirements of the Flight Capsule data output. These data will be analyzed by the mission operations personnel and published in a form suitable for presentation to members of the engineering and scientific teams to determine the success of the Capsule System mission.

4.0 FLIGHT CAPSULE PERFORMANCE AND DESIGN REQUIREMENTS

4.1 RELIABILITY

4.1.1 Probability of Mission Success

The Capsule System shall successfully complete its portion of the planetary exploration mission with a probability of 0.75 or greater.

4.1.2 Applicable Specification

Where not covered below, compliance shall be made to NPC-250-1 (Reliability Program Provisions for Space System Contractors, NASA: July 1963) in its entirety.

4.1.2.1 Circuit Protection

Electrical power shall be disconnected from squibs 0.5 seconds after firing to prevent battery drainage.

For each electrical circuit, consideration shall be given to inclusion of protective features (warm-up circuits, transient filters, and diodes to prevent shorts) and adaptive controls (such as drift and bias regulation using stabilizing local oscillators and other feedback loops).

4.1.2.2 Checkout and Calibration

The design of each functional unit shall incorporate an automatic test means, using part of an internally stored program, to verify operating status (or mode), performance range of the function, and to perform orderly and rapid diagnosis in event of failure.

All measuring instruments, sensors, and transducers shall be capable of remote calibration.

4.1.2.3 Circuit Initiation

To initiate any functional sequence, at least two independent signal sources shall be available and either signal alone shall be capable of sequence initiation.

To perform any functional sequence, two or more functionally independent (technologically different) means shall be available, whenever practical.

Where functionally independent means are impractical, two or more parallel (technologically identical) means shall be available to perform each functional sequence.

4.2 STERILIZATION

The planet quarantine constraint for Mars, which states that the probability of a viable microorganism existing on a capsule that enters the planet's atmosphere shall be less than one part in ten thousand, shall be satisfied by designing the Flight Capsule System, hardware, and procedures to comply with the following requirements and criteria:

1. All flight hardware that enters the planet's atmosphere shall be encapsulated in a container (sterilization canister) to provide biological isolation.

2. The Flight Capsule shall be sterilized by subjecting it to a terminal dry heat temperature cycle selected from the following table for required multiples of the specified death times that will satisfy the planetary quarantine. This shall be accomplished at the latest possible time prior to launch, compatible with prelaunch mission performance readiness tests and operations. The sterilization canister shall not be violated during such operations as are necessary to conduct the launch.

Acceptable Terminal Sterilization Cycles

Temperature (°C)	D Values* (hours)
160	0.21
155	0.31
150	0.46
145	0.73
140	1.1
135	1.8
130	2.8
125	4.4
120	7.0
115	11.0
110	17.5
105	28.0

*Acceptable exposure times to reduce a given viable population by 90 percent.

3. Opening of the canister subsequent to terminal heating, for repairs or any other reason, in other than a certified sterile environment shall be defined as a compromise to sterility and require a repeat of the terminal sterilization cycle and recertification of the capsule's sterility.

4. The canister shall not be deployed during mission operation until the risk of recontamination by terrestrial organisms is of a magnitude consistent with the quarantine constraint.

5. At the time of the terminal heat cycle, the viable biological population (burden) within the sterilization canister shall be certified to be at least 10^4 organisms less than the kill effectivity of the selected terminal heat cycle. Certification shall consist of adequate documentation of assay data obtained during the manufacture, assembly, and test of the capsule prior to terminal heat sterilization, consistent with the probability requirements of the overall quarantine constraint.

Chemical decontamination processes and procedures may be employed to reduce and control burden. If such processes and procedures are used, evidence of their burden reduction effectivity must be submitted, as obtained from an approved type approval program.

6. Every effort shall be made, consistent with other system design requirements, to minimize surface-to-surface contact areas and any other features which would tend to encapsulate and protect organisms against cleaning and decontamination processes and procedures.

7. Proof of sterilization by terminal heating shall be provided in the following manner:

a. During the application of the sterilization cycle, the oven temperature of the most thermally isolated parts of the system must be monitored. Recorded data must indicate that the most thermally isolated part of the system has received the required heat application.

b. Evidence that the heat cycle produces the degree of desired biological kill in the Flight Capsule must be obtained in the course of the program. Whether this evidence is obtained in development demonstration tests, type approval tests, or in conjunction with actual flight hardware is yet to be determined.

4.3 ENVIRONMENTAL

Paragraph 3.1.6 presents a summary of the environmental conditions expected to be encountered during a Capsule System mission. The Flight Capsule shall be designed to function within the limits of this specification after exposure to these environmental conditions.

4.4 ALIGNMENT AND ACCURACY

The Flight Capsule shall be designed within the following alignment and accuracy constraints.

1. The Separated Vehicle center of gravity location error shall not exceed 0.5 inch (3 sigma).

2. The de-orbit rocket thrust axis shall be located within 0.06 inch (3 sigma) of the Separated Vehicle centerline.

3. The rocket thrust vector angular misalignment shall be within 0.5 degrees (3 sigma) with respect to the Separated Vehicle centerline.

4. The Flight Spacecraft sensor error shall not exceed 0.16 degree (3 sigma) from the reference axes of the Flight Spacecraft.

5. Alignment of the Flight Capsule inertial reference subsystem shall be within 0.65 degrees (3 sigma) with respect to the Flight Spacecraft reference axes.

6. The thrust vector control and attitude control reaction nozzles shall be located at the extremity of the vehicle so as to produce a moment arm of 88 inches.

4.5 DIAGNOSTIC MEASUREMENTS

4.5.1 General

All subsystems and critical components shall be equipped with diagnostic outputs to the telecommunications subsystem which will provide the following:

1. Information to determine the cause of a mission failure.

2. Information to determine the cause of a subsystem failure.

3. Information to determine the cause of a critical component failure.

4. Information which is necessary for more accurate reduction of engineering data.

5. Information which will aid in determining the performance of subsystems and components.

4.5.2 Output Format

All voltage, current, temperature, pressure, vibration, and ablation outputs shall be 0 to 5 vdc analog. All separation, initiation and continuity outputs shall be in the form of "on/off" or binary data.

4.5.3 Measurements

The measurements to be made shall be in accordance with telecommunications data handling requirements in paragraph 6.11.2.

4.6 SAFETY

4.6.1 General

Factors that present a real or potential hazard to equipment and/or personnel typically include pyrotechnic devices, solid propellants, high-pressure gas-storage vessels, high voltages, toxic materials, and radioactive materials.

All systems shall consider all these, and any other equipment and personnel hazards, in the design of flight and ground equipment; and in the handling, use, and testing. Safety factors for pressure vessels, lines, and valves shall be considered, and the rationale for determining the tradeoffs between vehicle performance and personnel safety shall be documented. At the same time, the hazards to the flight equipment itself shall be considered. Consideration shall be given to safety techniques to be employed while testing and verifying flight hardware during times when pressure vessels must be partially or fully charged, squibs installed, radio-active sources used, etc. The design of hoisting, handling, and testing fixtures shall give special attention to minimize hazard to both personnel and equipment. The procedures utilized to fill pressure vessels, install and connect squibs, and load rocket propellants shall also take these safety aspects into consideration.

4.6.2 Flight Capsule Requirements

Some of the more specific requirements to comply with the above general requirements are as follows:

1. All pyrotechnics and propellants shall be stable during and after exposure to the environments listed in paragraph 3.1.6.

2. All pyrotechnic devices shall be safed by logic circuits requiring three independent functions in unique sequence to arm the devices. Ignition shall require an additional signal.

3. Pyrotechnics to be operated prior to entry of the Martian atmosphere shall not be armed until after launch.

4. Pyrotechnics to be operated after entry shall not be armed prior to completion of retro-thrust maneuver.

5. All squibs and igniters shall be short circuited until armed.

6. Squibs and igniters shall not fire with a current of 1 ampere passed through the terminals for a period of 1 minute.

7. Igniters shall be physically isolated from the rocket propellants until the Planetary Vehicle is out of the Earth's atmosphere.

8. All pressure vessels shall be equipped with diaphragm relief valves which shall rupture at a pressure differential between the maximum operating pressure and tank proof test pressure.

9. All pyrotechnic devices, charges, and rocket propellants shall be contained, positioned and located such that accidental initiation during ground operations precludes possibility of personnel injury, and premature initiation during the mission minimizes opportunity of hardware damage.

10. For each pyrotechnic device a fault tree diagram identifying all possible paths which could accidentally or prematurely fire the device is required. Numerical probability of an unintended firing for any device from all paths must be less than 1 chance in 1 million (10^{-6}).

4.6.3 Operational Support Equipment Requirements

4.6.3.1

General Range Safety Plan, Volume I and Associated Appendix A, AFETR P80-2 shall apply.

4.6.3.2

The OSE shall incorporate features to protect against damage to itself or the Flight Spacecraft caused by failure of the OSE, Planetary Vehicle, or the test facilities.

4.6.3.3

The automatic mode of testing shall provide sufficient safeguards in its programming to prevent the occurrence of damage due to improper sequencing.

4.6.3.4

The Planetary Vehicle shall be placed in a safe mode in case of test facility failure.

4.6.3.5

Temperature alarm monitoring shall be furnished in the OSE while the Planetary Vehicle or any of its subsystems are under thermal environmental test.

4.7 ELECTRICAL BONDING AND SHIELDING

4.7.1 Shielding

Equipment shielding shall afford a minimum design objective of 60 db attenuation to electromagnetic fields. Each equipment case shall be designed to afford maximum utilization of the structure for electrostatic and radio frequency shielding, whereby the equipment case forms a continuous metallic enclosure to the maximum extent possible. All necessary mechanical discontinuities in the case (such as inspection plates, covers, and joints) shall be electrically continuous across the interface of the discontinuity to provide the minimum radio frequency impedance current path. Hinges do not provide satisfactory low-impedance current path, and shall be treated as electrical discontinuities. Dependable prevention of electrical discontinuities require that the entire interface surfaces be in continuous intimate electrical contact along their entire length.

Recommended materials for maximizing electrical continuity include knitted wire gaskets and multiple spring-loaded contact strips. Where watertight or pressure-tight closure is required, combination gaskets of knitted wire and a sealing material such as neoprene or closed-cell sponge rubber are recommended. Mating interface surfaces shall be thoroughly clean of any oxide film or insulating finish. Conductive corrosion-resistant surfaces may be provided by Alodine or Iridite finish on aluminum, or other surfaces of equal conductivity. The use of anodized interface surfaces on aluminum, or direct contact between bare aluminum interface surfaces is expressly prohibited. Steel interface surfaces may be finished with tin, zinc, or cadmium, or other surfaces of equivalent conductivity and corrosion resistance. Direct contact of bare steel to bare aluminum, bare steel to bare magnesium, and bare aluminum to bare magnesium are prohibited. Permanent joints in equipment cases shall be formed by continuous seam welds or brazing. Spot welding or screw attachment shall be used only in conjunction with conductive metal gasketing or beryllium copper finger contact strips.

All conductors, including shields not directly bonded to the case, which penetrate the case and are capable of conducting interfering current either into or out of equipment, shall enter through suitable filters or feed-through capacitors with input and output terminals isolated by the case. However, the number of such components, shall be minimized. They should be used only where their necessity is positively indicated by analysis and test.

4.7.2 Component, Grounding, and Lead Configuration

4.7.2.1 Component and Lead Configuration

Components and wiring shall be placed and oriented to minimize harmful coupling among them, and to minimize the number of suppression components and amount of shielding required. Where there is conflict between orientation and placement requirements for radio-frequency protection versus dc and audio-frequency magnetic field protection, priority shall be given to the dc and audio-frequency magnetic field protection requirements.

4.7.2.2 Circuit Shielding Requirements

1. Audio Shields -- Audio shields shall be grounded at one point only. Each shield shall be insulated to maintain isolation from any chassis or structure ground at all points but one.

2. Radio Frequency (RF) Shields -- RF cable shield shall be grounded at a minimum of 2 points. RF cable shields shall make good electrical contact at both ends with the equipment enclosure, preferably by direct contact completely around the connector shells. Overall cable shields shall be defined as RF shields.

3. Conductor Shielding -- It is preferred that interference reduction be accomplished within the equipment when such means give results equal to or better than the use of conductor shielding. Line shielding on primary power leads to equipment is generally prohibited. However, due to the possible generation of high frequency harmonics by equipment such as high interference generators or inverters, further investigation and consideration shall be required as to whether primary power line shielding would be advantageous to the system.

Conductor shields shall be insulated from any contact with grounded conductors, chassis, or structure except at the required ground point or points. Conductor shields shall not be used for any single conductor (except for RF circuits). Shields shall be tightly woven copper and tinned copper braid with at least 90 percent coverage and shall be doubled where substantial shielding improvement can be demonstrated by so doing. Excessive conductor shielding shall be avoided for low-impedance audio frequency circuits where conductor separation, simple suppression, or twisting of conductors is more effective in reducing inductive coupling.

Shields shall not be generally employed as current conductors on any line designed to handle frequencies below 50 kilocycles. Individual return conductors shall not be shielded separately from "hot" conductors, but shall be included in the same shield with the "hot" conductor. Signal and power circuits operating below 50 kilocycles shall generally be carried in tightly twisted pairs wherever possible, and shielded if necessary. Circuits operating at 50 kilocycles or higher shall utilize adequately shielded coaxial cable or shall utilize balanced, shielded RF cable of lowest practical impedance.

4. Grounding Requirements -- The system design shall generally incorporate a single-point grounding scheme for all circuits and balanced circuit shields operating at less than 50 kilocycles. Multiple-point grounding shall be generally employed for circuits and shields operating at 50 kilocycles or higher, but precautions shall be taken to ensure that such grounding does not unintentionally add grounding points to the single-point ground system. This can be accomplished by using transformer coupling in ac systems or circuits, and transformer coupled ac-dc converters, for dc systems or circuits. A single ground point shall be established in the Flight Capsule. This point shall be referred to as vehicle ground point (VGP). All return ground leads shall be carried through any intermediate connectors, equipment, and cables to the VGP, and shall be insulated to prevent inadvertent contact with any other grounded conductor, chassis, or structure.

5. Return Ground Leads -- A separate conductor shall be carried from each stage junction point to VGP for each of the five types of return ground leads defined below. When return ground leads of a given type must be combined within an equipment case due to lack of space or connector contacts, a return bus shall be used. The return bus shall have minimum length and shall preferably be flat. The return bus cross section shall be at least as great as the combined cross sections of the return ground conductors connected to it.

6. Shield Return Ground Leads -- Shield return ground leads, except for RF shields, shall not intentionally carry any current, and may be combined wherever convenient to reduce the number of conductors and connector contacts required. When routing cable shields through connectors, the shields shall be carried through at least two cable pins.

RF cable shields shall be grounded on both sides of discontinuities. The ground leads shall be not more than 1 inch in length and shall be grounded to a surface free of finishes. Each shield shall be covered with an insulation to prevent interference from contact with other shields or structures. Two types of shield grounds are utilized:

a. Unbalanced critical signal circuits that are sensitive to RF interference shall have their shields grounded at each end.

b. Single-grounded shields (SGS) shall generally be used on all differential input or balanced circuits operating below 50 kc.

7. <u>Chassis Return Grounds</u> -- Chassis grounds shall be grounded to the Flight Capsule ground point via a chassis return ground lead.

8. <u>Return Ground Lead Types</u> -- Separate busses shall be installed in each equipment for the following:

a. AC return ground (power)

b. DC return ground (power)

c. Circuit return ground (isolated audio circuits)

d. Shield return ground (for single-grounded shields to be grounded at VGP).

e. Chassis return ground.

9. <u>Circuit Returns</u> -- Circuit returns shall not be shielded separately or carried outside the shield used for the corresponding "hot" conductor. Leads carrying audio frequency currents shall be tightly twisted with their returns and carried in insulated shields that shall generally be grounded at one end only in balanced or differential circuits operating below 50 kc.

10. <u>Power Returns</u> -- Power returns shall be routed with the "hot side" as a twisted pair to reduce the generation of magnetic fields, and shall be routed as close as possible to the structure. The power returns shall also pass through adjacent connector pins to reduce capacitive coupling to other circuits. All equipment power return leads shall be isolated from the equipment cases and grounded only at the VGP. This requirement is to reduce the possibility of any "ground loops" arising. Power circuits shall be kept as far from signal circuits as possible and shall cross other circuits at right angles where possible, to minimize inductive mutual coupling. This crossing shall apply to both the inside equipment cases and the external interconnecting cables.

11. Isolation Requirements -- There shall exist a minimum isolation of 1 megohm between the unit case and any point on all audio frequency signal or power circuits and a design goal of 100 megohms for high impedance audio-frequency-sensitive signal circuits.

12. Bonding -- The impedance of all electrical bonds shall be kept as low as possible. The impedance of any single bond shall not exceed 80 milliohms at any frequency below 1 mc. A single-bond impedance of 10 milliohms at any frequency below 1 mc shall be established as a design goal. The bonding between stages of the Space Vehicle shall be such that the impedance between any interface is 10 milliohms or less to minimize potentials between modules.

4.7.3 Impedance of Audio Frequency Circuits

Interconnecting line load termination impedance shall not be less than 50 ohms and between 150 and 600 ohms except for those special requirements of high impedance instrumentation circuits. The line source impedance shall be restricted. Equipment interconnections shall be designed so that for each interconnecting circuit the signal current load and signal return current lead (equal and opposite currents) are routed together in the same bundle. Individual interconnecting circuits shall pass through connectors via adjacent pins. Internal wiring design shall utilize wire twisting and wire routing control so as to minimize the formation of magnetic field pick-up loops. Shields on interconnecting wires used to exclude electric fields shall be carried through connectors for ultimate grounding at a single point ground and shall be covered by a layer of insulation.

4.7.4 RF Hazard to Electro-Explosives Devices

Protection shall be provided for all electro-explosive devices to maximize safety against premature initiation by stray electrical energy in any form. This requires assurance that any electro-explosive device will not absorb sufficient power for ignition when the vehicle is exposed to system power transients, high-level RF fields coupling to other circuits, or electrostatic charges. The levels of exposure shall be determined by analysis of transient and radio frequency field data and radio frequency field data from tests on the Space Vehicle and estimates of radiation power expected in the operational environment.

4.7.5 Electrical Power Systems Requirements

4.7.5.1 AC Power System

Alternating current power leads shall be separated from communication, control, and instrumentation circuits as far as practicable in order to minimize interference caused by power wires. Their routing within the vehicle is to be conducted in accordance with standard wire separation methods. All power leads with their respective neutral leads are to be twisted together to form separate cables to cancel the magnetic and electric field that exists around the wires.

4.7.5.2 DC Power System

The dc power system shall be a 28 volt, 2-wire system. In order to minimize the susceptibility of the direct current systems to magnetic and electric field disturbances, their respective positive and negative leads are to be tightly twisted.

4.7.5.3 Electrical Equipment Design

All return (neutral) ground wires shall return to the neutral bus and shall be of the same size and material as their counterpart "hot wires". Their circuitry shall be isolated from the electrical equipments metallic housing and grounded at the neutral bus only.

4.7.5.4 Electronic Circuit Design

All signal and control wires shall be tightly twisted with their current return wires for their entire length to form individual electronic circuits.

All electronic circuits shall, where practicable, be terminated in their respective characteristic impedance to enhance transmission quality and reduce disturbing effects on other circuits.

All electronic circuits shall be separated from power circuits as far as practicable. Their separation distance may vary within the module but they are not to be routed in the same conduit run. The lower the signal level and the more sensitive the electronic circuit, the greater is the required separation distance.

4.7.6 Transmission Lines

Equipment requiring antennas, but not employing waveguide or coaxial cable, shall be designed to utilize shielded transmission lines. Receiving antennas, or any other low-level signal circuits, shall be low impedance, or of balanced design, so that coaxial or other shielded transmission lines can be used to insure an interference-free installation. The receiving antenna input or any low-level signal circuit input within the equipment shall be kept to a minimum. Antenna or low-level signal circuit return paths or ground paths shall be so arranged that minimum interference will result due to common conductive paths with other circuits or with the enclosing case grounding path.

4.8 ELECTROMAGNETIC INTERFERENCE

4.8.1 General

The Capsule System and all components and subsystems making up the system shall be designed in accordance with MIL-I-26600 (USAF) Military Specification Interference Control Requirements, Aeronautical Equipment,

4.8.2 Specific

4.8.2.1 Each assembly in the Flight Capsule shall be compatible with respect to other assemblies therein, to other equipment in the Flight Spacecraft and to the Launch Vehicle.

4.8.2.2 Interference suppression components shall be used only where inherent suppression is not practical. When filters are required, they shall be made integral with the equipment.

4.8.2.3 Switching transients shall be suppressed by utilizing RC networds, RF bypass capacitors, diodes, etc.

4.8.2.4 Susceptibility to fields and voltages from other circuits, components, and equipments shall be minimized to the greatest extent practical in the basic design of each component, assembly and subsystem.

4.8.2.5 The equipment shall be designed to withstand, without degrading the operation, the transient supply voltage changes caused by operation of other equipment in the system.

4.8.2.6 The system shall be designed to withstand, without degrading, the operation, the maximum RF voltages found to exist as a result of operation of the following:

1. Radar or radio transmitters at frequencies within the design timing range of the transmitters.

2. Spurious and harmonic transmitter frequencies

3. Radiation from local oscillators

4. Audio-frequency fields

5. Radiated and conducted interference from properly suppressed electrical and electronic equipment installed anywhere in the system.

4.8.2.7 All digital or audio pulse-circuits shall be designed to operate at as high a triggering voltage level as feasible, definitely above the

the millivolt level. This design will provide optimum freedom from inadvertant operation due to stray pulses.

4.9 MAGNETIC CLEANLINESS

4.9.1 General

The Flight Capsule shall be designed so that its functional operations shall minimize the generation of a magnetic field. Particular emphasis is placed upon selection of materials and routing of electrical circuitry.

4.9.2 Requirements

4.9.2.1 Allowable magnetic fields

1. The total magnetic field from dc through 5.0 cps of an assembly or subassembly shall not exceed 5.0 gamma measured at 3.0 feet.

2. Magnetic field above a frequency of 5.0 cps are of no significance.

3. The current-loop contribution to the Flight Spacecraft magnetic field shall at no time exceed 0.5 gamma measured at 3.0 feet from the current-carrying conductor.

4.9.2.2 Stability

The total magnetic field of an assembly or subassembly shall not change by more than 0.5 gamma measured at 3.0 feet after exposure to the mission environments.

4.9.2.3 Materials

Nonmagnetic materials shall be used wherever possible.

4.9.2.4 Magnetic Components

If it is necessary to use components or assemblies which are magnetic (e.g., iron-core relays, amplitrons, and magnetrons), the field of these magnets shall be minimized; preferential or back-to-back layout is recommended. Increasing the efficiency of the magnetic circuit within a component will decrease the radiated magnetic field and thereby increase reliability and decrease weight.

4.9.2.5 Wiring

In general, the current path and its ground return shall be closely adjacent conductors.

4.9.2.6 Shielding

Magnetic Shielding shall be minimized wherever possible.

4.9.2.7 Shielding constraints

1. Shielding is more effective if the item shielded is small (largest dimension is less than 4.5 inches) and if the stability properties of the shield are better than the item shielded. Shields with longest dimension greater than 4.5 inches require special consideration.

2. Shielding shall be placed only around the component as a whole. Discontinuities (e.g., sharp bends, corners, holes) seriously degrade the shielding effectiveness and shall be avoided.

3. Shielding may be required where the item to be shielded contains a magnet or where the field of the item may change magnitude or direction (e.g., rotating parts).

4.10 MAINTENANCE

The design shall be capable of withstanding up to 6 months storage and 500 hours of checkout and test with no maintenance other than battery charging and conditioning. Specific maintenance and repair cycles will be identified later.

4.11 USEFUL LIFE

After assembly and sterilization the Flight Capsule is required to survive 6 months storage, 500 hours of test and checkout, 2 hours of launch operations and Earth transfer orbit, 234 days of interplanetary transit, and up to 3 hours of operation.

4.12 TRANSPORTABILITY

All equipment must be capable of withstanding the transportation environments listed in Figure 4.

4.13 DESIGN AND CONSTRUCTION STANDARDS

4.13.1 Selection of Specification and Standards

The selection of specifications and standards shall be developed at a later date using this specification as a basis. All standards and specifications shall be considered in addition to those contractually established and approved by the procuring agency.

4.13.2 Materials and Processes

Material and process requirements shall be developed during the design using governmental specifications and standards where applicable and creating contractor specifications where necessary.

4.13.3 Standard and Commercial Parts

Primary choice of parts, materials, and circuits shall come from approved lists to be developed for reliability, sterilizability, and compatibility with environments wherever available. Choice from nonapproved sources shall be accomplished by evidence of acceptability or by plans for tests which can produce such evidence.

4.13.4 Moisture and Fungus Resistance

All parts are to be moisture proof and fungus inert and must be capable of passing environmental tests specified in MIL-STD-810 or be protected from these environments by coating or packaging.

4.13.5 Corrosion of Metal Parts

Metal parts shall be protected from corrosion by coatings or treatments, to be determined as detail design and material choices are made. Adjacent parts shall be compatible from an electrolytic standpoint or be coated or plated to prevent electrolytic corrosion.

4.13.6 Interchangeability and Replaceability

All parts shall be configuration controlled and standardized where possible to allow interchange and replacement when necessary. The degree of control is to be determined.

4.13.7 Workmanship

All parts shall be fabricated and finished in such a manner that criteria of appearance, fit, and adherence to specific tolerances can be observed. Particular attention shall be given to the neatness and thoroughness of all welding, marking, plating, machine screw assembly, and freedom of parts from burrs and sharp edges.

Detail description of good workmanship for particular parts shall be established.

4.13.8 Identification and Marking

Detail identification and marking of parts shall be designated on part drawings and shall conform to MIL-STD-130.

4.14.9 Storage

Storage criteria shall be developed consonant with environments likely to be experienced by a presterilized but clean Flight Capsule as well as a sterilized and pressurized unit prior to launch.

4.13.10 Sterilization

The Flight Capsule shall be designed to be capable of terminal heat sterilization techniques which assure the probability of contamination of the planet shall not exceed 10^{-4}.

4.14 WEIGHT

The weight of the Flight Capsule on board the Flight Spacecraft shall not exceed 3000 pounds including whatever portion remains with the Flight Spacecraft after separation.

4.15 SCHEDULE

Since the mission objectives involve the 1971 Mars opportunity, Flight Capsule designs, techniques and components shall be compatible with the schedule. On this basis, the Flight Capsule design technology cutoff date for the 1971 mission shall be taken as September 1966.

4.16 TRAJECTORY

4.16.1 Definitions

4.16.1.1 A Launch opportunity is a reoccurring duration of time, every 25.6 months when favorable Earth - Mars spatial positions allow for practical interplanetary transfer trajectories.

4.16.1.2 A launch period is the number of days within the launch opportunity when practical Earth - Mars transfer trajectories are selected depending on mission objectives and Launch Vehicle constraints.

4.16.1.3 A launch window is the duration of time each Earth day when Space Vehicle launch is practical to achieve desired Planetary Vehicle transfer orbit orientation and characteristics depending on mission objectives and Launch Vehicle constraints.

4.16.2 Selected Design Conditions

The Flight Capsule design shall be compatible with the following Planetary Vehicle trajectory conditions:

1.	Launch opportunity	1971
2.	Launch period	10 May 1971 to 29 June 1971
3.	Launch window	2 hours minimum
4.	ZAP angle	70 to 76 degrees
5.	Communications range	176×10^6 km maximum
6.	Approach velocity	2.85 to 3.13 km/sec
7.	Departure velocity	2.80 to 4.24 km/sec
8.	Time of flight	184 to 234 days
9.	Arrival date	30 December 1971

5.0 CAPSULE SYSTEM DEFINITION

5.1 FLIGHT CAPSULE TERMINOLOGY

Figure 5 presents a further breakdown of the Flight Capsule, identifying the terminology at the operational stages of separation and/or deployment. In summary, the Flight Capsule is attached to the Flight Spacecraft by the forward and aft sections of the Flight Capsule to Flight Spacecraft adapter. Operation of the sterilization canister lid separation mechanism followed by the operation of the separation subsystem on the Flight Capsule to Flight Spacecraft adapter, results in the Separated Vehicle. Attitude control, propulsion, and thrust vectorial control maneuvers are performed to de-orbit the Separated Vehicle and place it on a preselected plantary impact trajectory. After the propellants have been expelled to perform these maneuvers, the resultant Entry Vehicle cruises to and enters the planet atmosphere. After entry, the entry shell (including the ACS and TVC reaction subsystems) is separated and the Suspended Capsule descends through the atmosphere with the parachute. Appropriate portions of the programming and sequencing, thermal control, telecommunications, electrical power and control, and the signal and power interconnection subsystems hardware are individually attached and operationally integrated to accomplish the data acquisition and transmission functions of the Capsule System mission.

5.2 PRIMARY FUNCTIONAL AREA LIST - INCLUDING COMPONENTS

The Flight Capsule shall be subdivided into eleven primary functional areas including major components as listed below. The components listed in each primary functional area may also be utilized in another area to accomplish another function of the Capsule System mission.

5.2.1 <u>Sterility Control</u>

1. Sterilization canister structure
2. Pressurization subsystem

5.2.2 <u>Separation</u>

1. None

5.2.3 <u>Programming and Sequencing</u>

1. Pressure switch -- reaction control manifold
2. Pressure switch -- sterilization canister

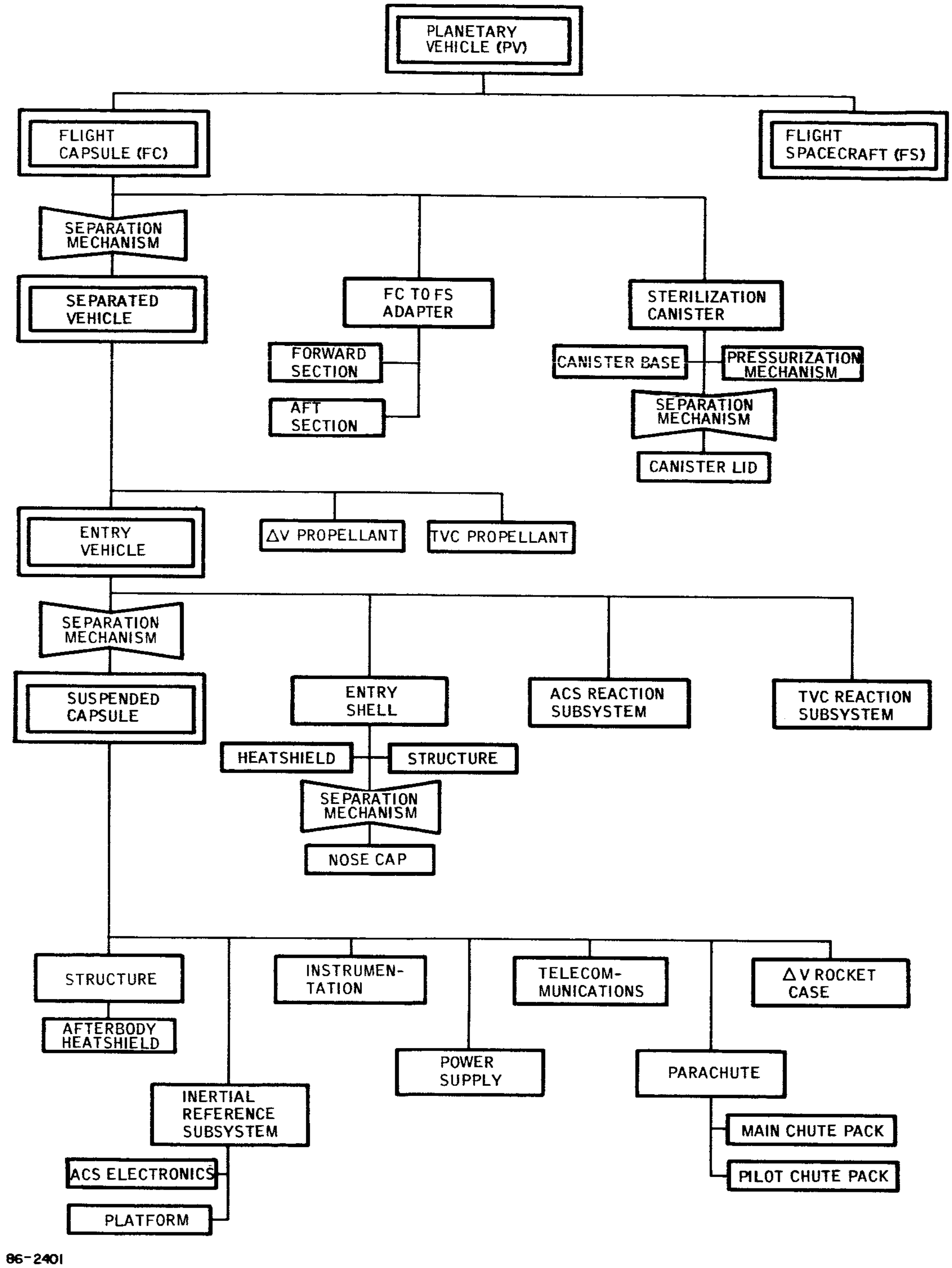

Figure 5 ENTRY FROM ORBIT FLIGHT CAPSULE - OPERATION DIAGRAM AND TERMINOLOGY

3. Separation switch -- sterilization canister lid

4. Separation switch -- Flight Capsule to Flight Spacecraft separation

5. Separation switch -- entry shell nose cap

6. Load switch -- parachute

7. Accelerometer -- X-axis

8. Altimeter -- high altitude

9. Altimeter -- low altitude

10. Limit switch -- camera platform

11. Electronic package -- stable platform

12. Central computer and sequencer

5.2.4 De-Orbit

1. Rocket engine

2. Exhaust nozzle extension

5.2.5 Attitude Control

1. Tank cold gas

2. Valve -- shut-off

3. Valve -- nozzle

4. Electronics package

5. Inertial reference subsystem

6. Sentry gyro

7. TVC gas generator

8. Valves and nozzles

5.2.6 Descent Retardation - Entry

1. Entry shell structure

2. Ablative heat shield

5.2.7 Descent Retardation - Postentry

1. Main parachute

2. Swivel

3. Harness assembly and attachments

4. Main parachute deployment bag

5. Main parachute canister

6. Gas-generator bag

7. Gas-generator propellant container

8. Pilot parachute

9. Pilot parachute mortar assembly

5.2.8 Thermal Control - Passive

1. Primary Heat Shield

   a. Ablator
   b. Bond
   c. Substructure

2. Secondary Heat Shield

   a. Ablator
   b. Bond
   c. Substructure

3. Afterbody Heat Shield

   a. Ablator
   b. Bond
   c. Substructure

4. Miscellaneous Subsystem Protective Shields

5.2.9 Data Acquisition - Engineering

1. Radiation detector

    a. Low-energy electron detectors
    b. High-energy electron detector
    c. Proton detector

2. Three accelerometers

3. Mass spectrometer

4. Acoustic densitometer

5. Gas chromatograph

6. Two-pressure gages

7. Two-temperature probes

8. Beta scattering instrument

9. Water detector

10. Television experiment

11. Penetrometer experiment

    a. Four penetrometers
    b. Penetrometer receiver

12. Radar altimeter experiment

    a. Transceiver
    b. High-altitude antenna
    c. Low-altitude antenna

13. Velocity-attitude sensor experiment

    a. Transceiver
    b. Two antennas

5.2.10 Electrical Power and Control

1. Two battery chargers

2. Two batteries

3. Two power converter regulators

4. Power control unit

5.2.11 Telecommunications

1. Flight Capsule Radio Subsystem

   a. Two antennas
   b. Two transmitters
   c. Two directional couples
   d. Two 3-port circulator switches

2. Flight Capsule Data Handling Subsystem

   a. Two experiment data handling subsystems
   b. Two diagnostic data handling subsystems
   c. Two delay data storage subsystems
   d. Two television data storage subsystems

3. Flight Spacecraft radio subsystem

   a. Two antennas
   b. Two receivers

4. Flight Spacecraft Data Handling Subsystem

   a. Two buffer data storage units
   b. Two bulk data storage units

5.3 FUNCTIONAL FLOW DIAGRAM

Figure 6 presents a functional flow diagram of a representative Flight Capsule as defined by the subsystem descriptions of the primary functional areas presented in Section 6.

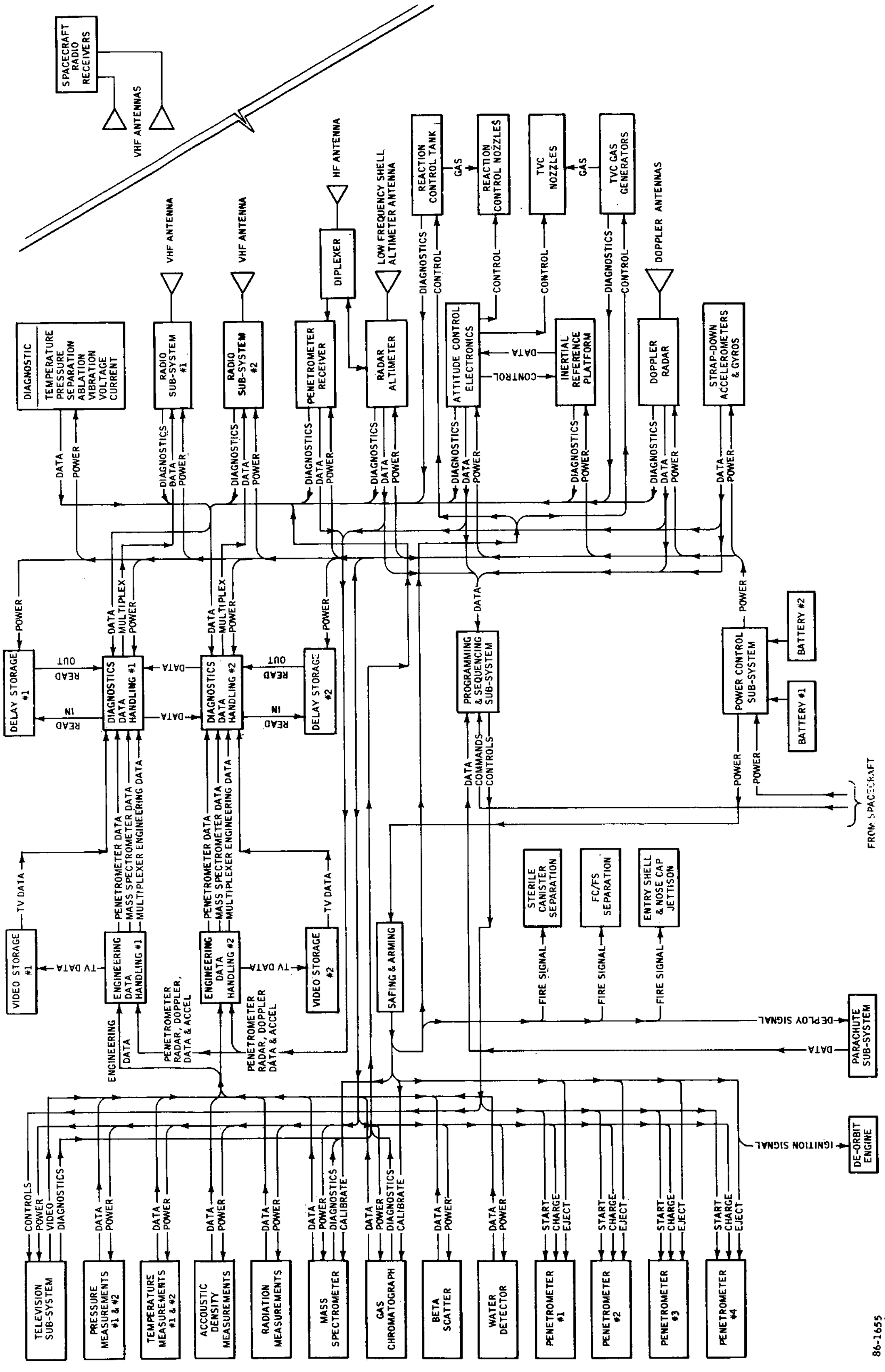

Figure 6 FUNCTIONAL FLOW DIAGRAM

5.4 INTERFACE REQUIREMENTS

The interface requirements defined herein shall be restricted to those which shall exist between the Capsule System and other systems involved in the project. These interfaces are defined from manufacture of the Flight Capsule through final data reduction and evaluation. They will not, however, define the many interfaces which shall exist within the Flight Capsule.

5.4.1 <u>Flight Capsule/Flight Spacecraft Interfaces</u>

5.4.1.1 Envelope

The Flight Spacecraft shall be capable of accepting a Flight Capsule with an envelope as defined in Figure 3.

5.4.1.2 Mounting Pattern

The Flight Capsule to Flight Spacecraft mounting adapter at the field joint shall have an 120-inch bolt circle.

5.4.1.3 Alignment

The Flight Capsule to Flight Spacecraft alignment accuracy shall be in accordance with paragraph 4.4.

5.4.1.4 Structural

Each Flight Spacecraft shall be capable of supporting the Flight Capsule weight of 3000 pounds during preflight, interplanetary injection, orbit injection, and separation maneuvers.

5.4.1.5 Center of Gravity (cg), Moments of Inertia (MI), Products of Inertia (PI) (as shown in Section 5.6.2).

5.4.1.6 Umbilical Connections

The Flight Spacecraft shall be equipped with umbilical connectors to both the Flight Capsule and to the Launch Vehicle. These connectors shall be capable of handling the signals, power, and communications as required. The information supplied through these umbilicals will be used during ground checkout to feed signals, power and control functions from the OSE through the Flight Spacecraft to the Flight Capsule. In addition to ground operations, the Flight Spacecraft to Flight Capsule umbilical will be utilized during interplanetary cruise in support of the Flight Capsule.

5.4.1.7 Power

The Flight Spacecraft shall be capable of supplying power to the Flight Capsule as shown in Table II. In addition, power from the OSE shall be routed through the Flight Spacecraft during checkouts while the vehicles are mated.

5.4.1.8 Signals

The Flight Spacecraft shall be capable of receiving commands from the Operational Support Equipment and Deep Space Network as well as generating commands in its own Central Computer and Sequencer (CC&S) and transmitting these commands via hardlines to the Flight Capsule. These commands shall be in accordance with the requirements specified in paragraph 6.3. Control functions from the OSE shall be routed through the Flight Spacecraft during checkouts while the vehicles are mated.

5.4.1.9 Communications

The Flight Spacecraft shall be capable of accepting telemetry data from the Flight Capsule, via either hard line through the umbilicals or via radio link, for storage and retransmission to the DSN. The Flight Spacecraft shall also receive command information and convert it to a form suitable for use by the Flight Capsule.

5.4.1.10 Radio Interference

The Flight Spacecraft and the Flight Capsule shall comply with paragraph 4.8 of this specification.

5.4.1.11 Grounding and Bonding

The Flight Spacecraft shall utilize a single point ground system for all low frequency functions from the Flight Capsule. The mechanical joint between the Flight Capsule and the Flight Spacecraft shall be such as to provide unified behavior when the Planetary Vehicle is excited by a RF field.

5.4.1.12 Software

Flight Capsule interface definition document requirements include but are not limited to the following mission operations: scheduling, trajectory, arrival geometry, orbit geometry, lifetime, operating sequence, communications requirements, data acquisition and handling requirements, postlaunch decisions, and logic.

TABLE II

FLIGHT SPACECRAFT TO FLIGHT CAPSULE POWER REQUIREMENTS

Function	Phase	Volts	Amperes	Watts	Duration
F C System Operation	1. Launch to Interplanetary cruise	F C operates on internal power			
	2. Preseparation	30 ± 10	20	481(1)	3-1/2 hours
Thermal control	1. Interplanetary cruise	*	*	*	Continuous
	2. Warm up to operating temperature prior to each checkout	*	*	*	0-24 hours
Battery charge	1. After checkouts during interplanetary cruise	30-40	low	low	As required
	2. After checkout preseparation	30-40	≈ 0.7	24	3-1/2 hours
Gyro warmup	1. Prior to each checkout	30 ± 10	≈ 0.3	10	1 hour

(1) Additional power required for pyrotechnics, thermal control, etc. shall be determined.

5.4.2 Launch Vehicle/Flight Capsule Interfaces

5.4.2.1 Mechanical

The Launch Vehicle shall be capable of inserting two Planetary Vehicles into the interplanetary trajectory. The weight of the Flight Capsule portion of the Planetory Vehicle shall not exceed 3000 pounds. Adequate dynamic clearance between the Flight Capsule and the Launch Vehicle ascent fairing and instrument compartment shall be maintained throughout all phases of Launch Vehicle operation.

5.4.2.2 Umbilical Connections

The Launch Vehicle shall have umbilical connectors which mate with both the Launch Complex Equipment (LCE) and the Planetary Vehicle which are properly interconnected to permit operation of the Flight Capsule from the launch complex including indicators, controls, and monitors.

5.4.2.3 Radio Interference

The Launch Vehicle shall conform to MIL-I-26600, "Interference Requirements, Aeronautical Equipment" plus applicable NASA specifications.

5.4.2.4 Grounding and Bonding

All grounding and bonding of cable shields through the Launch Vehicle shall be in accordance with a single point grounding philosophy. In general, shields will be carried through umbilical connectors and not connected to the airframe. In instances where shields cannot be carried through connectors, these shields will be connected to airframe ground as close to the connector as possible, but connected to the airframe ground at one point only.

5.4.2.5 Scheduling and Performance

Launch Vehicle interfaces with the Capsule System will include scheduling and performance capability.

5.4.3 Operational Support Equipment/Flight Capsule Interfaces

5.4.3.1 Electrical

1. Systems Test Complex (STC) -- The OSE designed for use in the STC shall be capable of functioning in the Flight Capsule for complete electrical checkout and also for final acceptance tests. This equipment shall be usable on a subsystem level as well as on a system level for these tests. The OSE shall supply such functions as mode control, command control, timer updating, external power, battery conditioning, monitor loops, RF and hard line readouts, etc. This equipment shall be designed such that no pyrotechnic or propulsion devices can be operated.

2. Launch Complex Equipment (LCE) -- The LCE shall supply the Flight Capsule with the same functions as the STC with the exception that it operates on a system level only.

3. Flight Capsule Simulators -- The OSE shall simulate devices which are electrically identical to the Flight Capsule at its interfaces. These devices shall be used in compatibility testing between the Flight Spacecraft and the Flight Capsule, the Planetary Vehicle and Launch Vehicle and between the Space Vehicle and the Launch Complex Equipment.

4. Mission Dependent Equipment (MDE) -- The MDE used in conjunction with the DSN shall be compatible with DSIF, and the Flight Capsule and/or Planetary Vehicle information format, so as to be capable of providing automatic readout of the Flight Capsule and/or Planetary Vehicle data during all phases of the mission.

5.4.3.2 Mechanical

1. Handling -- The OSE must be capable of providing the Flight Capsule with suitable stands, jigs, fixtures, cradles, cranes, etc., for handling and positioning the Flight Capsule for various tests. These tests will include center of gravity (cg), moment of inertia (MI), product of inertia (PI), fit checks, alignment checks, mating checks and final mating with Planetary Vehicle mounted on the Launch Vehicle.

2. Transportation -- The OSE shall provide shipping containers capable of maintaining the environmental conditions specified during handling and transporting the Flight Capsule from one facility to another.

3. Test and Checkout -- The OSE shall provide fixtures which can be used in performing fit and alignment acceptance tests.

4. Safety -- The OSE shall be designed to provide the maximum safety to personnel and equipment during all handling, checkout, mating, and launch phases of the mission. Specific areas which must be considered are pyrotechnic devices, high pressure vessels, high voltage devices, grounding and bonding, etc. An Explosive Safe Facility (ESF) shall be provided at KSC for propellant and pyrotechnic installation, and storage.

5. Software -- Interfaces between the Capsule System and the OSE will consist of developing schedules, test plans, test procedures, handling procedures, safety procedures, and test requirements.

6. Thermal -- During transportation and storage the OSE shipping and storage containers shall keep the temperature experienced by the Flight Capsule within the range of 35 to 125°F.

5.4.4 Deep Space Network/Flight Capsule Interfaces

5.4.4.1 Deep Space Instrumentation Facility (DSIF)

The DSIF shall be capable of transmitting commands to and receiving Flight Spacecraft telemetry data from the Flight Capsule, in combination with the Mission Dependent Equipment. The DSIF shall also be capable of providing the MOS with range and tracking information for up-dating timers, selecting backup modes, etc.

5.4.4.2 Deep Space Operations Facility

The DSOF shall be capable of accepting data from the DSIF, interpreting this data, and supplying the DSIF with appropriate information for command through the Flight Spacecraft to the Flight Capsule.

5.4.5 Kennedy Space Center (KSC)/Flight Capsule Interfaces

The major interface between the Capsule System and KSC will be scheduling and mission planning. A plan must be developed to provide a suitable daily launch window over the specified launch period. During these periods, all communications, tracking, and data acquisition facilities must be available for the program. In addition, KSC shall provide storage, laboratory, assembly, test and administrative areas for the Capsule System.

5.5 EVENT SEQUENCE

The Flight Capsule event sequence shall be considered to begin at such time as all systems have been integrated into the Space Vehicle and are ready to launch. The sequence shall be considered complete when a Suspended Capsule has impacted the planet surface and no further communications from it are possible.

Table III presents the Flight Capsule checkout sequence for both the prelaunch and interplanetary transit phases. The discrete command, DC-6, which initiates this sequence, can be issued by the LCE through the Flight Spacecraft for prelaunch operations or by the MOS through the Flight Spacecraft for flight operations.

Table IV presents a jettison sterile canister lid sequence initiated by the discrete command DC-4 from the MOS to the Flight Spacecraft CC and S via the Flight Spacecraft command receiver.

Table V presents the balance of the Flight Spacecraft events from preseparation to impact on the surface of the planet initiated by discrete command DC-1.

The abbreviations used in Tables III, IV and V are listed below:

C - Time of checkout (prelaunch or interplanetary transit)

L - Time of canister-lid separation

S - Time of Flight Spacecraft - Flight Capsule separation

M - Time of Flight Spacecraft attitude maneuver

E - Time of entry into Martian atmosphere

D - Time of parachute deployment

H - Time of first penetrometer ejection

TABLE III

EVENT SEQUENCE - PRELAUNCH OR INTERPLANETARY TRANSIT CHECKOUT

Event	Time	Form	Destination	Remarks
				Checkout sequence initiated by discrete command DC-6
1. Switch thermal control to "Operate"	C-t_1 hours	Pulse	Thermal control	To warm all components to operating temperature. Time t_1 is a preset number which will be determined
2. Switch gyros on external power	C-60 minutes	Pulse	ACS	To stabilize gyros
3a. Switch all systems on-FC power	C	Pulse	Power control	
3b. Initiate subsystems electrical calibration	C	Pulse	Calibrator	To check ACS and accelerometers
4a. Electrical calibration	C+2 minutes	Pulse	Calibration	
4b. Open gas samples	C+2 minutes	Pulse	Gas chromatograph	Opens gas flask to calibrate chromatograph, mass spectrometer and densitometer (This event is inhibited unless discrete command DC 7 accompanies DC 6)
4c. Uncage stable platform	C+2 minutes	Pulse	Platform integrators	To measure drift
5a. Illuminate camera lenses	C+25 minutes	Pulse	TV cameras	
5b. Take TV pictures	C+25 minutes	Pulse	TV cameras	To check camera status
6a. Cage platform	C+30 minutes	Pulse	Power Control	
6b. Switch all systems off	C+30 minutes	Pulse	Power control	
6c. Trickle charge batteries from Flight Spacecraft power	C+30 minutes	Pulse	Power Control	Replace energy used in checkout
6d. Switch thermal control	C+30 minutes	Pulse	Power Control	

TABLE IV

EVENT SEQUENCE-JETTISON STERILE CANISTER LID

Event	Time	Form	Destination	Remarks
				Discrete command DC4 from Flight Spacecraft to CC&S starts sequence-
1. Arm S&A device	L-5 minutes	Pulse	Sterile canister S&A	Pulse from Flight Spacecraft arms S&A device
2. Jettison sterile canister lid	L	Pulse	Lid pyro	Command inhibited unless pressure in canister is less than 0.02 psia (canister should have been depressurized in Earth orbit) END JETTISON LID EVENTS

TABLE V

EVENT SEQUENCE - PRESEPARATION TO IMPACT

Event	Time	Form	Destination	Remarks
				Master Sequence initiated by discrete command DC1
1. Switch thermal control to "Operate"	S-t_1 hours	Pulse	Thermal control	To warm all components to operating temperature - time t_1 is a preset number
2. Switch gyros on - FS power	S-302 minutes	Pulse	ACS	To stabilize gyros
3a. Swith all systems on - FC power	S-242 minutes	Pulse	Power control	
3b. Start subsystem electrical calibration	S-242 minutes	Pulse	Calibrator	To check ACS and accelerometers
4a. Electrical calibration complete	S-240 minutes	Pulse	Calibrator	
4b. Open gas samples	S-240 minutes	Pulse	Gas chroma-graph	Opens gas flask to calibrate chromatograph, mass spectro-meter and densitometer
4c. Uncage stable platform	S-240 minutes	Pulse	Platform integrators	To measure drift: if drift exceeds a preset quantity, separation is inhibited
5a. Illuminate camera lenses	S-215 minutes	Pulse	TV cameras	
5b. Take TV pictures	S-215 minutes	Pulse	TV cameras	
6a. Cage platform	S-210 minutes	Pulse	TV cameras	
6b. Switch all systems to FS power	S-210 minutes	Pulse	Power control	
6c. Charge batteries from FS power	S-210 minutes	Pulse		Replace energy used in checkout
7a. Stop battery charge	S-2 minutes	Pulse	Power control	
7b. Switch FS systems to FC power	S-2 minutes	Pulse	Power control	As a backup to command 7b, FS automatically switches to FC power when separation switch closes
8a. Arm separation pyros	S-1 minute	Pulse	S&A device	
8b. Open cold gas supply	S-1 minute	Pulse	Power control	Pyrotechnic valve - separation inhibited unless reaction control is pressurized
9. Uncage platform	S-1 second	Pulse	ACS	
10. Separate FC from FS	S	Pulse	Power control	Inhibited until sterilization can lid separation switches are open - inhibited until separation enable command (DC 3) received
11. Enable ACS system	S+5 seconds	Pulse	ACS	ACS disabled until S+5 sec to prevent bumping
12. Command roll angle	S+3 minutes	Digital pulse train	ACS electronics	Stored command QC1 transferred to ACS electronics

TABLE V (Cont'd)

EVENT SEQUENCE - PRESEPARATION TO IMPACT

Event	Time	Form	Destination	Remarks
13. Command pitch angle	S+8 minutes	Digital pulse train		Inhibited unless roll angle confirmed - stored command QC2 transferred to AC electronics
14. Enable TV subsystem	S+40 minutes	Pulse	ACS electronics	Propulsion not initiated until FC has separated 1 km from FS and roll and pitch angles have been confirmed
15. Ignite propulsion rocket	S+40 minutes	Signal internal to CC&S	Rocket igniters	
16. Shut off TVC electronics	S+45 minutes	Signal internal to CC&S	TVC	To save power
17. Command roll angle	M	Digital pulse train	ACS	Start maneuver time M is stored command QC5: Up to 3 maneuvers may be used - roll angle is stored command QC 6
18. Command pitch angle	M+5 minutes	Digital pulse train	ACS	Inhibited until roll angle confirmed pitch angle is stored, command QC 7
19. Arm entry pyrotechnics	E-15 minutes	Pulse	S&I device	
20a. Activate Mach no. computer	E-3 minutes	Internal signal	CC&S	
20b. Deactivate pitch and yaw control, maintain roll control	E-3 minutes	Pulse	ACS	Reaction control electronics and stable platform left on until impact to maintain entry vehicle roll control to parachute deployment
21a. Deploy pilot parachute	D	Pulse	Chute container	When Mach number $\leq$ 1.2 and altitude <27000 feet <u>or</u> when Mach number $\leq$0.85 <u>or</u> when altitude $\leq$20,000 feet
21b. Deactivate reaction control	D	Pulse		
22. Main chute deployment	D+2 seconds		Main chute container	Backup to pilot chute deployment gas generator forces main chute out of container 2 seconds after pilot chute deployment signal is generated
23a. Cut entry shell attachments 23b. Energize doppler radar 23c. High altitude altimeter "off" 23d. Low altitude altimeter "on"	See Remarks		Attachment pyros	Command given when load switch on parachute riser senses load or when accelerometer indicates 5ge increase in deceleration in 0.5 second interval

TABLE V (Concl'd)

EVENT SEQUENCE - PRESEPARATION TO IMPACT

Event	Time	Form	Destination	Remarks
24. Start TV sequence	D+3 to D+11.2 seconds		TV camera shutter	First TV pictures taken between 3 and 11.2 seconds after chute deployment and as rapidly as they can be broadcast thereafter
25a. Eject first penetrometer	H		Penetrometer ejector	Penetrometer ejection sequence initiated as function of altitude H or time from maximum g_e
25b. Change data mode	H	Pulse	Data handling	
26. Eject second penetrometer	H+2 seconds	Pulse	Penetrometer ejector	Command given 2 seconds after first penetrometer is ejected
27. Eject third penetrometer	H+4 seconds	Pulse	Penetrometer ejector	Command given 4 seconds after first penetrometer is ejected
28. Eject fourth penetrometer	H+6 seconds	Pulse	Penetrometer ejector	Command given 6 seconds after first penetrometer is ejected
29. Impact - End of mission				

5.6 CONFIGURATION

5.6.1 Design Description

The inboard profiles of the required Flight Capsule design are presented in Figures 7 and 8 in the launch and entry configurations respectively. The Flight Capsule shall utilize the blunt cone (60-degrees slope) as the basic configuration, as shown in Figure 9, made up of several major assemblies; sterilization canister, adapter, support structure, and entry shell.

5.6.1.1 Sterilization Canister

This assembly shall be constructed of a thin shell that completely encloses the Separated Vehicle including a portion of the Flight Spacecraft to Flight Capsule adapter. Local access areas shall be provided for assembly and handling. A pressurization system shall be utilized to maintain a slight pressure differential across the canister from sterilization through launch into the Earth parking orbit.

5.6.1.2 Flight Spacecraft to Flight Capsule

This adapter shall create the mounting and load path support system to the Flight Capsule during launch and interplanetary travel. It shall be constructed of a conical shell with mounting rings at either end. This adapter shall also house the Separated Vehicle separation mechanism, located at the Separated Vehicle/Flight Spacecraft to Flight Capsule adapter interconnection.

5.6.1.3 Suspended Capsule Structure

The structure shall serve several purposes:

1. Provide a load path system for the complete separated vehicle during ΔV thrusting and for parachute loads.

2. Form the mounting surface for all of the payload and parachute system.

3. Provide the mounting and separation functions for the entry shell.

The basic structure shall be made up of eight beams running radially from the ΔV propulsion unit to the entry shell interface as shown in Figure 10. The outer surface of the beams shall be tied together with a conical shell and the inner surface of the beams shall be tied together with a cylindrical shell.

5.6.1.4 Entry Shell

The entry shell shall be constructed of a thermal protection system, a primary structure and a secondary structure. The thermal protection system shall utilize an ablative material as both the primary heatshield (forward side of primary structure) and secondary heat shield (aft side of primary structure).

5.6.2 <u>Weight and Mass Properties</u>

The weight summary and mass properties presented in Tables VI and VII respectively are representative of the design shown in Figures 7 and 8.

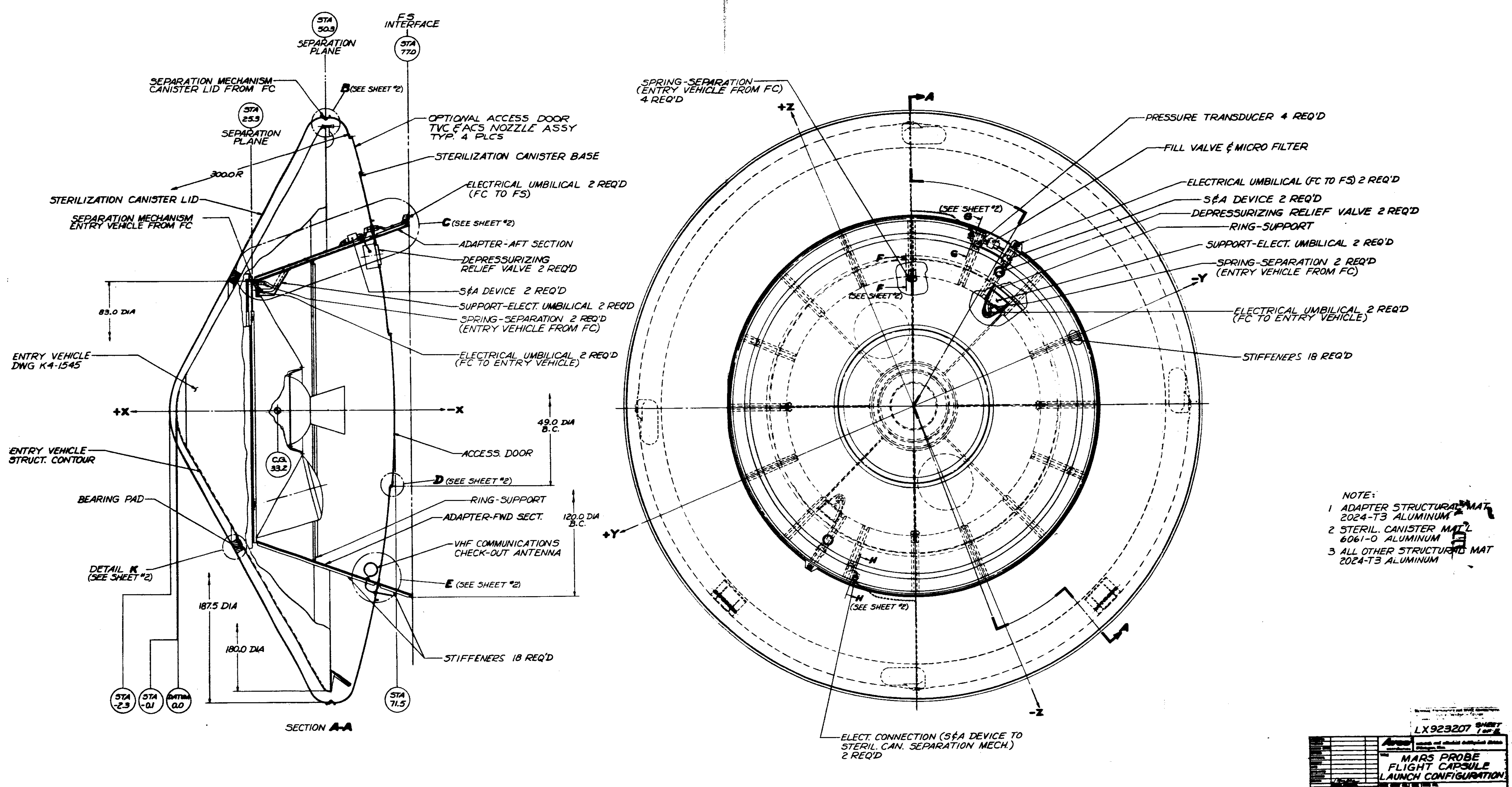

Figure 7 FLIGHT CAPSULE - LAUNCH CONFIGURATION

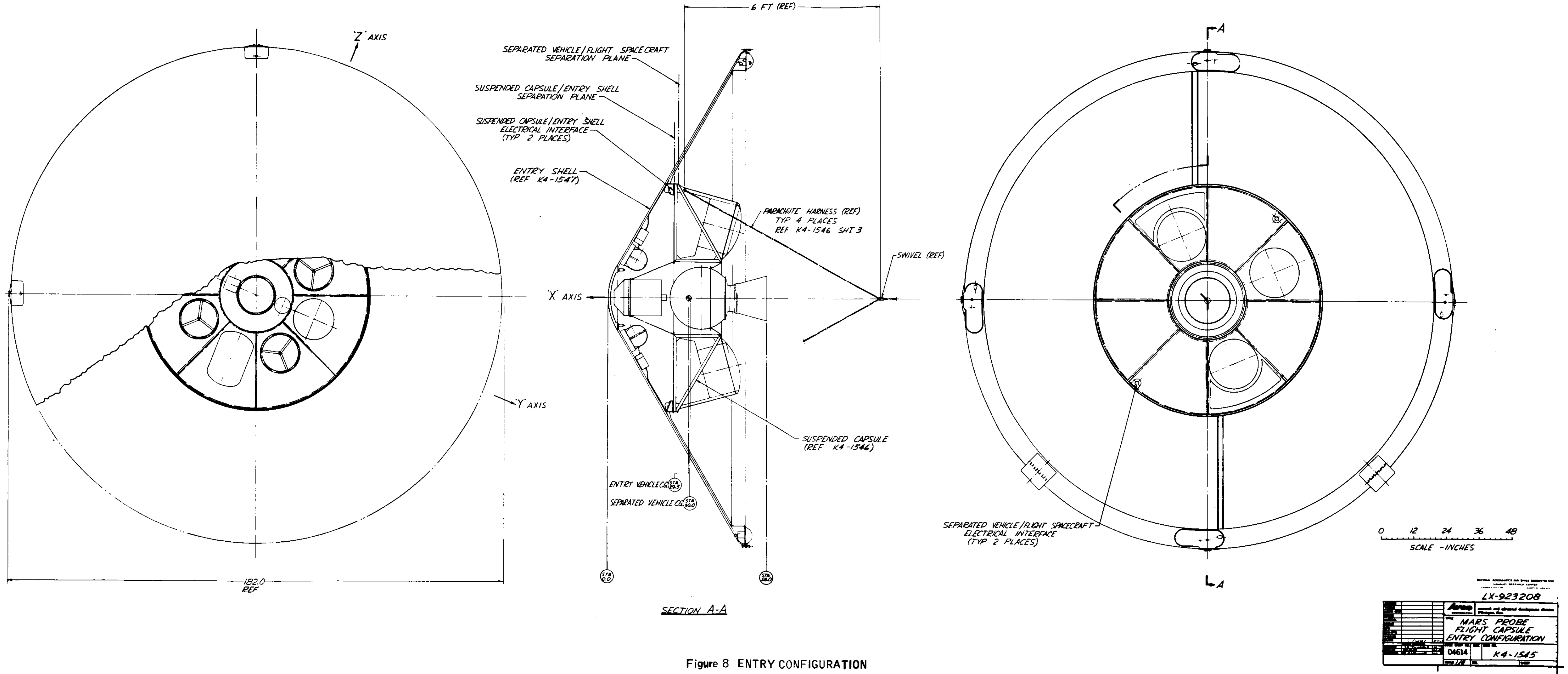

Figure 8 ENTRY CONFIGURATION

70-1

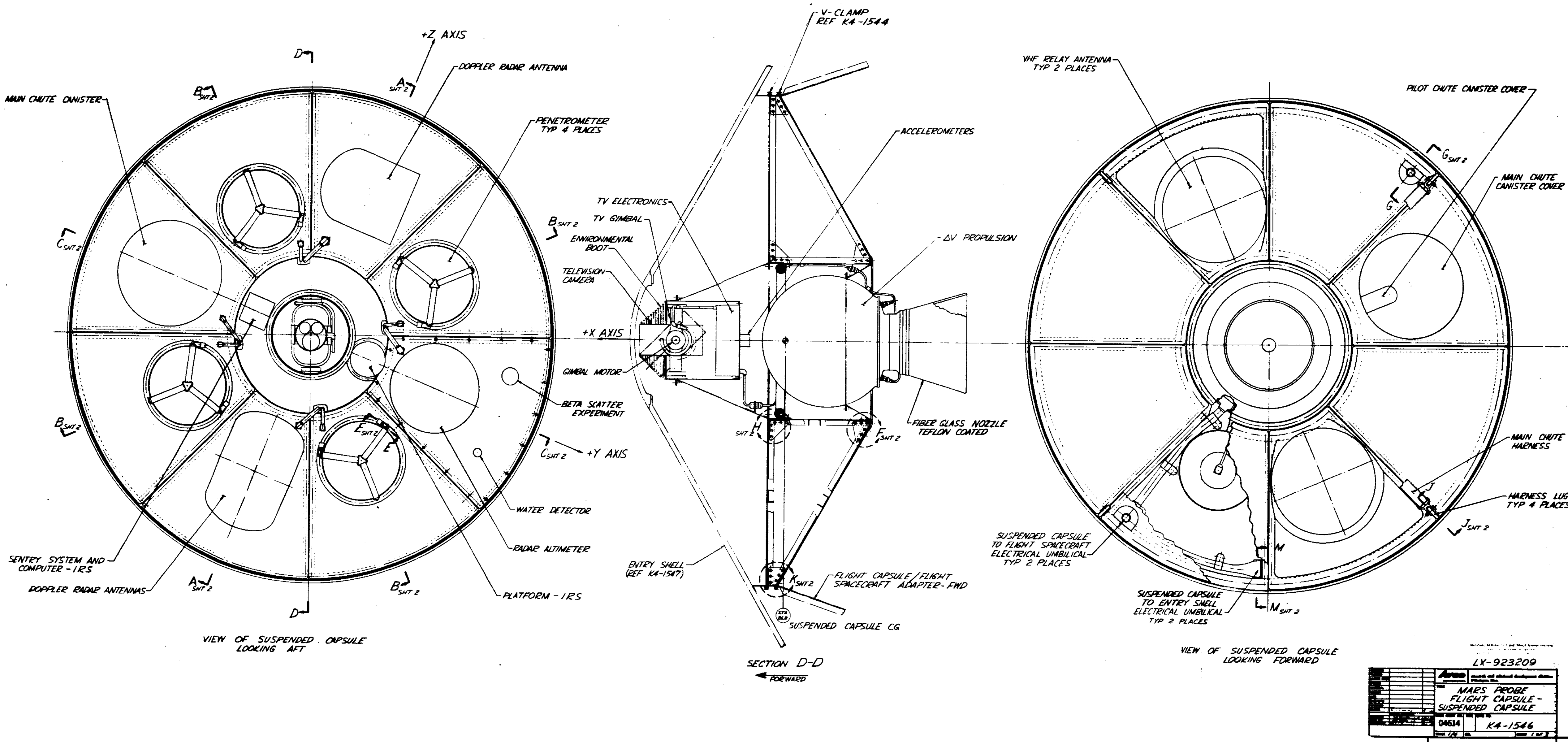

Figure 9 ENTRY SHELL

71-1

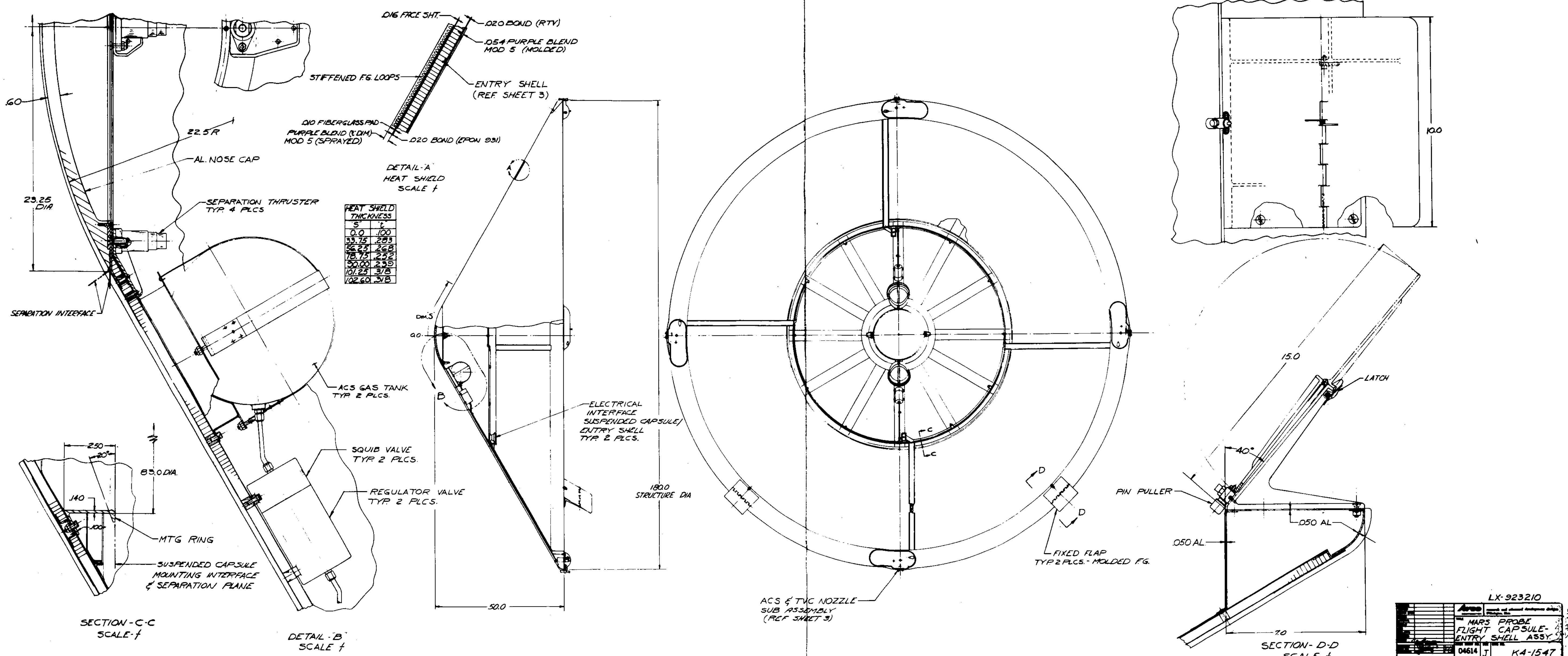

Figure 10 SUSPENDED CAPSULE

72-1

-72-2

TABLE VI

WEIGHT SUMMARY
$M/C_DA = 0.22\ slug/ft^2$
Diameter - 15 feet

Flight Capsule		2922.1
Sterilization canister lid	125.0	
Pressurization gas	15.0	
Pre-FC Separation		2872.1
Sterile canister base	163.0	
Pressurization nozzle valves	6.0	
FC - FS adapter	125.0	
Hardware, brackets, cables	29.5	
Separated Vehicle		2458.6
Propulsion propellant	400.0	
ACS gas expelled	1.0	
FVC gas expelled	17.6	
Entry Vehicle		2040.0
Entry shell heat shield	370.7	
Entry shell structure	343.0	
Thermal control	30.0	
ACS - reaction control	42.4	
TVC - reaction control	48.5	
Hardware, brackets, cables	83.5	
Available for growth	96.9	
Suspended Capsule		1025.0
Instrumentation	205.6	
Radar	56.9	
Telecommunications	117.4	
Power supply	178.0	
Parachute	84.0	
Initial reference subsystem	21.6	
Propulsion case	49.0	
Structure	96.0	
Hardware, brackets, cables	131.0	
Programming and sequencing	23.6	
Afterbody heat shield	36.0	
Available for growth	25.9	

TABLE VII

MASS PROPERTIES SUMMARY

	Total Weight (pounds)	I_{xx} (slug-ft^2)	I_{yy} (slug-ft^2)	cg_x (in. from nose)
Suspended Capsule	1025	131	97	26.8
Entry Vehicle	2040	1046	575	29.5
Separated Vehicle	2459	1042	581	30.0
Pre FC Separation	3782	1262	720	33.2
Flight Capsule	2922	1362	780	33.2

Note: I_{xy} and I_{xz} cross products shall not exceed 0.1.

6.0 FLIGHT CAPSULE PRIMARY FUNCTIONAL AREAS

6.1 STERILITY CONTROL

6.1.1 Functions

The sterilization canister shall maintain the required sterile environment and provide a continuous surface for passive thermal control of the Separated Vehicle from the end of terminal sterilization until the canister lid is jettisoned.

6.1.2 Performance/Design Requirements

6.1.2.1 Probability of Mars Contamination

The probability of Mars contamination by a single Flight Capsule shall not exceed 10^{-4}.

6.1.2.2 Sterilization Cycle

The sterilization canister assembly shall be able to function after exposure to a sterilization cycle and environments of paragraphs 4.3 and 4.4.

6.1.2.3 Internal Pressure

The internal pressure shall always be greater than the ambient pressure to inhibit the entrance of microorganisms through minute leaks from the time of terminal sterilization through launch into an Earth transfer orbit.

6.1.2.4 Venting

The canister shall be vented during ascent. Venting shall maintain pressure slightly above ambient until Planetary Vehicle is injected into Earth orbit, at which time the pressure shall drop to ambient.

6.1.2.5 Separation

The canister lid separation shall be accomplished without contaminating or damaging adjacent equipment on the Flight Capsule or Flight Spacecraft.

6.1.2.6 Surface Emissivity

1. Inside:

Lid - $\epsilon < 0.10$
Base - $\epsilon < 0.05$

2. Outside:

Lid - $\epsilon < 0.05$
Base - $\epsilon < 0.05$

6.1.3 Functional Interfaces

6.1.3.1 Electrical

1. Separation Connectors

No inflight separation connectors needed.

2. Control Signals Required

a. Shut off valve--prior to auxiliary tank disconnect

b. Pressure dump valve--at lift off

c. Lid separation--see separation requirements in paragraph 6.2.1

3. Signal Sources

a. Shut off valve--FC LCE

b. Pressure dump valve--FS CC&S

c. Lid separation circuit--FC programmer

4. Initiating Current:

4.5 ampere dc minimum

6.1.3.2 Mechanical

1. Pressurization subsystem fill valve and plumbing connection used during sterilization cycle and prelaunch storage.

6.1.3.3 Thermal

1. Conduction from canister base to canister lid to entry heat shield.

2. Conduction from Flight Spacecraft through Flight Spacecraft Flight Capsule adapter.

3. Radiation from Flight Spacecraft.

4. Radiation to space.

6.1.4 Component List

1. Canister structure

2. Pressurization subsystem

3. Separation subsystem

6.2 SEPARATION

A separation function shall be required for each of the following operations of the Flight Capsule mission:

1. Sterilization canister lid separation
2. Separation Vehicle separation
3. Parachute ejection
4. Entry shell separation
5. Nose cap separation
6. Penetrometer separation

6.2.1 Sterilization Canister Lid

6.2.1.1 Function

Cut the biologically sealed canister and force the lid away from the expected path of the Entry Vehicle when subsequently separated from the Flight Spacecraft.

6.2.1.2 Performance/Design Requirements

The canister lid shall be detached and separated by an explosively actuated mechanism with the following requirements:

1. Impulse

Minimum impulse applied to lid for separation is 5.59 lb-sec.

2. Ordnance No Fire Characteristics

a. 1.0 ampere for 5 minutes

b. 1.0 watt for 5 minutes with 1.0 volt applied

3. Actuating Current

4.5 amps

4. Initiating Circuit Resistance

1.5 ohms maximum

5. Actuating Time

20 msec maximum

6. Interference

The canister lid shall not interfere with the Separated Vehicle or Spacecraft.

7. Redundancy

Redundant detonators and detonator igniter circuits shall be provided.

8. Safety

Detonators to be physically isolated from the separation mechanism until after the launch phase.

9. Contamination

All parts and debris shall be retained.

6.2.1.3 Functional Interfaces

1. Electrical

a. Initiating Current--4.5 amp dc minimum

b. Signal Source--Signal required from safing and initiating circuit (dual circuit for redundancy).

c. Electrical Disconnect--Not required

d. Safety--Igniter circuits to be isolated from power by a barometric switch until after launch.

2. Mechanical

a. Weight--Maximum separated lid weight: 125 pounds.

b. Leakage--Leakage shall be minimized up to venting in Earth during launch.

3. Thermal

a. Maximum sterilization temperature 275°F

b. Minimum operating temperature - 100°F

6.2.1.4 Component List

1. Mild Detonating Fuse

2. Detonator

6.2.2 Separated Vehicle

6.2.2.1 Function

Separate the Separated Vehicle from the Flight Spacecraft with sufficient velocity to obtain adequate spacing to prevent impingment of the subsequently initiated ΔV rocket engine exhaust plume on the Flight Spacecraft.

6.2.2.2 Performance/Design Requirements

The Separated Vehicle shall be disconnected from the Flight Spacecraft to Flight Capsule adapter section by a "V" type clamp ring released by any one of four explosive nuts and separated by springs. Separation of the electrical umbilical shall be accomplished in combination with the above separation function.

1. Velocity of Separation

1.5 ft/sec minimum

2. <u>Ordnance No-Fire Characteristics</u>

   a. 1.0 amp for 5 minutes

   b. 1.0 watt for 5 minutes with 1.0 volt applied

3. <u>Actuating Current</u>

   4.5 amp dc

4. <u>Actuating Time</u>

   20 msec maximum

5. <u>Tip off rate (maximum)</u>

   0.3 deg/sec

6. <u>Initiating Circuit Resistance</u>

   1.5 ohms maximum

7. <u>Redundancy</u>

   Dual igniter circuits on each of four nuts; release of any one nut releases clamp ring.

8. <u>Debris</u>

   All parts of clamp ring shall be retained on the Flight Capsule to Flight Spacecraft adapter.

6.2.2.3 Functional Interfaces

1. <u>Electrical</u>

   a. Initiating Current - 4.5 amps dc minimum

   b. Signal Source - Signal required from safing and initiating circuits (dual circuit for redundancy).

   c. Electrical Disconnect - No signal required.

d. Safety - Igniter circuit to be isolated from power by barometric switch until after launch.

2. Mechanical

a. Mates forward Flight Spacecraft to Flight Capsule adapter to Suspended Capsule

b. Ejected by eight springs (retain springs on Flight Spacecraft to Flight Capsule adapter)

c. Releases electrical connection to Flight Spacecraft

3. Thermal

a. Maximum temperature 300°F

b. Minimum temperature to be determined.

6.2.2.4 Components

1. V Clamp

2. Explosive Nuts (or separator)

3. Power Cartridge

6.2.3 Parachute

6.2.3.1 Function

Eject the pilot parachute into the gas stream to deploy the main parachute.

6.2.3.2 Performance/Design Requirements

A pilot parachute shall be propelled into the gas stream by a mortar which in turn pulls the main parachute from its storage canister into the gas stream.

1. Pilot Parachute Ejection Velocity

100 ft/sec.

2. Ordnance No-Fire Characteristics

a. 1.0 amp for 5 minutes.

b. 1.0 watt for 5 minutes with 1.0 volts applied.

3. Actuating Current

4.5 amps dc

4. Initiating Circuit Resistance

1.5 ohm maximum

5. Redundancy

Redundancy shall be provided in ejection mechanisms, explosive actuators, and explosive igniter circuits.

6. Debris

All small parts and explosive debris shall be retained.

6.2.3.3 Functional Interfaces

1. Electrical

a. Initiating Current -- 4.5 amps dc minimum

b. Signal Source -- Signal required from safing initiating circuits (dual circuits for redundancy).

c. Electrical disconnect -- None required

d. Safety -- Electric circuit to be isolated from power by barametric switch until after launch.

2. Mechanical

Separation mechanism is internal to parachute pack.

6.2.3.4 Component List

1. Parachute Pack Assembly

6.2.4 Entry Shell

6.2.4.1 Function

Detach the entry shell after peak load of parachute deployment to reduce the weight of the Suspended Capsule and allow deployment of instruments.

6.2.4.2 Performance/Design Requirements

A "V" type clamp ring shall release the entry shell by the firing of any one, or all, of four explosive nuts. Separation of the electrical umbilical shall be accomplished in combination with the above separation function.

1. Separation Timing

   Separation must be timed to avoid entry shell impact on the Suspended Capsule due to parachute induced dynamics.

2. Ordnance No-Fire Characteristics

   a. 1.0 amp for 5 minutes

   b. 1.0 watt for 5 minutes with 1.0 volt applied

3. Actuating Current

   4.5 amp dc

4. Actuating Time

   20 msec maximum

5. Initiating Circuit Resistance

   1.5 ohm maximum

6. Redundancy

   Dual pressure cartridges for each nut; dual igniter circuits in each cartridge

7. Debris

   All parts of clamp ring shall be retained with the entry shell.

6.2.4.3 Functional Interfaces

1. Electrical

a. Initiating Current -- 4.5 amp dc minimum

b. Signal Source -- Signal required from parachute load cell or accelerometer indication of parachute opening.

c. Electrical Disconnect -- No signal required

d. Safety -- Igniter circuit to be isolated from power until separation of the parachute.

2. Mechanical

Mates forward ring of Suspended Capsule structure with mounting ring of entry shell.

6.2.4.4 Component List

1. "V" Clamp

2. Explosive Nuts (or separator)

3. Power Cartridge

6.2.5 Nose Cap

6.2.5.1 Function

Separate and eject the nose cap from the entry shell to allow access for the TV camera if the entry shell fails to separate.

6.2.5.1 Performance/Design Requirements

The nose cap shall be released and ejected by four thruster bolts at parachute deployment if entry shell fails to separate.

1. Impulse

52 lb-sec minimum applied to nose cap for separation

2. Ordnance No-Fire Characteristics

a. 1.0 amp for 5 minutes

b. 1.0 watt for 5 minutes with 1.0 volt applied

3. Actuating Current

4.5 amps dc

4. Initiating Circuit Resistance

   1.5 ohms maximum

5. Redundancy

   Dual pressure cartridges for each bolt and dual igniter circuits for each cartridge.

6. Ejection Angle

   5.0 degrees to vehicle axis

7. Debris

   All small parts and gaseous products shall be retained.

6.2.5.3 Functional Interfaces

1. Electrical

   a. Initiating Current -- 4.5 amps dc minimum

   b. Signal Source -- Signal required from safing and and initiating circuits (dual circuits for redundancy).

   c. Electrical Disconnect -- None required

   d. Safety -- Electrical circuits shall be isolated from power by barometric switch until after launch.

2. Mechanical

   Nose cap shall be the spherical portion of entry shell appropriately mated to minimize leakage prior to separation.

3. Thermal

   a. Maximum temperature -- 425°F

   b. Minimum temperature -- to be determined

6.2.5.4 Component List

1. Thruster Bolts

2. Power Cartridge

6.2.6 Penetrometer

6.2.6.1 Function

Release penetrometers to measure surface hardness of planet.

6.2.6.2 Performance/Design Requirements

Explosive pinpullers release the retaining straps and the penetrometer is spring ejected. Four penetrometers are released. The first one at 3500 feet above the surface of the planet and the remainder, one at a time at two second intervals.

1. Spring Impulse

   Maximum velocity 5 ft/sec

2. Ordnance No-Fire Characteristics

   a. 1.0 amp for 5 minutes

   b. 1.0 watts for 5 minutes with 1.0 volt applied

3. Actuating Current

   4.5 amps dc

4. Initiating Circuit Resistance

   1.5 amp dc minimum

5. Redundancy

   Dual pressure cartridges for each pinpuller and dual igniter circuits for each cartridge

6. Debris

   All explosive debris and small parts shall be retained.

6.2.6.3 Functional Interfaces

1. Electrical

   a. Initiating currents -- 4.5 amp dc minimum

   b. Signal Source -- Signal required from programmer (dual circuits for redundancy).

c. Signal Time

1) First Signal -- 3500 feet altitude to release one penetrometer

2) Second Signal -- 3500 feet altitude + 2 seconds to release second penetrometer

3) Third Signal -- 3500 feet altitude + 4 seconds to to release third penetrometer

4) Fourth Signal -- 3500 feet altitude + 6 seconds to release fourth penetrometer

d. Electrical Disconnect -- Cable cutter at each penetrometer

2. Mechanical

Pin pullers to hold two legs of a three leg retaining strap so the released strap shall not obstruct television.

3. Thermal

.a. Maximum temperature -- To be determined

b. Minimum temperature -- -100°F

6.2.6.4 Component List

1. Exlosive Nut

2. Power Cartridge

6.3 PROGRAMMING AND SEQUENCING

6.3.1 Functions

The programming and sequencing functional area shall perform the function of safing, timing, sequencing, and computational operations for the Flight Capsule.

The safing function shall prevent premature initiation or operation of any component or device where such initiation or operation might be hazardous to personnel and/or the success to the mission.

The timing function shall provide a time base from which all sequences and some computations are made.

The sequencing function shall initiate various events in proper order and time.

The computational operations shall solve various equations involving time, acceleration, and altitude in order to initiate various discrete events.

6.3.2 Performance/Design Requirements

6.3.2.1 Programming and Sequencing

Timing, sequencing, and computational operations shall be performed by the central computer and sequencer (CC&S). The CC&S shall be supplied electrical energy from either or both of two redundant power sources feeding the CC&S through two separate power lines. The CC&S shall contain a forward biased diode in each power input circuit in accordance with Figure 14.

The CC&S shall be capable of initiation discrete commands and sequences of events described in the following paragraphs. The same abbreviations used in the event sequence in paragraph 5.5 shall apply.

1. Checkout Sequence -- The checkout sequence shall control events necessary to determine condition and operating status of all subsystems as shown in Table VIII.

The sequence shall consist of 6 primary events occurring at fixed times following the receipt of discrete command (DC) 6 (start checkout sequence).

The clock controlling the checkout sequence shall have a 1 minute resolution.

2. Electrical Simulation Sequence -- The electrical simulation sequence shall provide a series of calibration signals to each of the accelerometers and gyros as shown in Table IX.

The accelerometer outputs provide both an instrument calibration and inputs to that portion of the CC&S which controls entry events and therefore permits exercising that logic. The gyro output signals cause the ACS electronics to operate the reaction control valves in response to the simulated gyro-output signals. The operation of the valves indicate proper operation of the ACS system.

The objective lenses of the television cameras shall be illuminated and one set of photographs shall be taken.

TABLE VIII

CHECKOUT SEQUENCE

Events	Time	Form	Destination	Remarks
				Checkout sequence initiated by discrete command (DC) 6
1. Switch thermal control to operate	C-t_1 hour	Pulse	Thermal control	To warm all components to operating temperature - time t_1 is a preset number which will be determined.
2. Switch gyros on - FS power	C-60 minutes	Pulse	ACS	To stabilize gyros
3a. Switch all systems on FC power	C	Pulse	Power control	
3b. Initiate electrical sequence stimulation	C	Pulse	Calibrator	To check ACS and accelerometers
4a. Electrical stimulation sequence complete	C+2 minutes	Pulse	Calibrator	
4b. Open gas samples	C+2 minutes	Pulse	Gas chromatograph	Opens gas flask to calibrate chromatograph, mass spectrometer and densitometer. This event is inhibited unless DC 7 accompanies DC 6.
4c. Uncage stable platform	C+2 minutes	Pulse	Stable platform integrators	To measure drift
5a. Illuminate camera lenses	C+25 minutes	Pulse	TV cameras	
5b. Take TV pictures	C+25 minutes		TV cameras	To check camera status
6a. Cage platform	C+30 minutes	Pulse	Stable platform	
6b. Switch all systems off	C+30 minutes	Pulse	Power control	
6c. Trickle charge batteries from FS power	C+30 minutes	Pulse	Power control	Replace energy used in checkout
6d. Switch thermal control to <u>nonoperate</u>	C+30 minutes	Pulse	Power control	End checkout events

TABLE IX

ELECTRICAL STIMULATION SEQUENCE

Events	Time	Form	Destination	Remarks
				Sequence initiated by event 3b of checkout or master sequence
1. Apply 0 percent calibration	T	Grounded circuit	Accelerometers	Both strapped down and stable platform accelerometers calibrated simultaneously
2. Apply 25 percent calibration	T+10	Direct current		
3. Apply 50 percent calibration	T+20			
4. Apply 75 percent calibration	T+30			
5. Apply 95 percent calibration	T+40			
6. Apply 0 percent calibration	T+50		Accelerometers	Linear decrease with time to initiate peak-g determination and Mach number computation
7. Apply + roll current	T+50		Gyro torque Motor	Stable platform, cold gas reaction system and TV camera platform operation checked simultaneously
8. Apply - roll current	T+60			
9. Apply + yaw current	T+70			
10. Apply - yaw current	T+80			
11. Apply + pitch current	T+90			
12. Apply - pitch current	T+100			
13. Pitch current off	T+110			
14. Energize high altitude altimeter in calibrate mode	T	Pulse	Altimeter	
15. Deenergize high altitude altimeter	T+50	Pulse		
16. Energize low altitude altimeter in calibrate mode	T+50	Pulse		
17. Deenergize low altitude altimeter	T+110	Pulse	Altimeter	END CALIBRATION SEQUENCE EVENTS

The reticle pattern in each picture indicates that each television camera is operating.

Calibration of the gas chromatograph mass spectrometer and acoustical densitometer shall be accomplished by releasing a sample of known gas to each instrument. Release of gas shall be inhibited unless DC 7 accompanies DC 6.

Calibration of both high and low altitude radar shall be determined by means of a self contained calibration net work with a fixed delay.

The clock controlling the electrical stimulation shall have a 5 second resolution.

3. Master Sequence -- The master sequence shall control all events from the start of the final checkout and calibration through entry. The sequence shall consist of 16 primary events occurring at predetermined times following the receipt of DC 1 (start master sequence), as shown on Table X.

The first five functions shall consist of the checkout sequence including release of gas for calibration.

At the completion of the checkout sequence all systems shall be switched to Flight Spacecraft power at which time the Flight Capsule batteries can start to be recharged.

If drift of the stable platform exceeds a predetermined limit, separation shall be inhibited.

Separation shall be inhibited until the reaction control manifold is pressurized.

When the Flight Capsule separates from the Flight Spacecraft, all Flight Capsule subsystems shall be turned on by means of a separation switch if, due to some malfunction, they had not previously been turned on.

4. Separation Sequence -- The separation sequence shall be controlled by a separate 1 second resolution clock. The separation sequence is shown on Table XI.

Separation shall be inhibited unless the sterilization canister lid separation switches are open and DC 3 (separation enable) has been received.

TABLE X

MASTER SEQUENCE

Events	Time	Form	Destination	Remarks
				Master sequence initiated by DC 1
1. Switch thermal control to operate	S-t_1 hour	Pulse	Thermal control	To warm all components to operating temperature. Time t_1 is a preset number.
2. Switch gyros on - FS power	S-302 minutes		ACS	To stabilize gyros
3a. Switch all subsystems on FC power	S-242 minutes		Power control	
3b. Electrical stimulation sequence-on	S-242 minutes		Calibrator	To check ACS and accelerometers
4a. Electrical stimulation sequence-off	S-240 minutes		Calibrator	
4b. Open gas samples	S-240 minutes		Gas chromatograph	Opens gas flask to calibrate chromatograph, mass spectrometer and densitometer
4c. Uncage stable platform	S-240 minutes		Platform integrators	To measure drift: If drift exceeds a preset quantity separation is inhibited
5a. Illuminate camera	S-215 minutes		TV cameras	
5b. Take TV pictures	S-215 minutes		TV cameras	
6a. Cage platform	S-210 minutes		Platform integrators	
6b. Switch all subsystems to FS power	S-210 minutes		Power control	
6c. Charge batteries from FS power	S-210 minutes		Power control	Replace energy used in checkout
7a. Stop battery charge	S-2 minutes		Power control	
7b. Switch FS subsystems to FC power	S-2 minutes		Power control	As a backup to command 7b, FC automatically switches to FC power when separation switch closes
8a. Arm separation pyros	S-1 minute		S & A Device	
8b. Open cold gas supply			Power control	Pyrotechnic valve. Separation inhibited unless reaction control is pressureized
8c. Initiate separation Sequence Clock	S-1 minute	Signal internal to CC &S	Separation clock in CC & S	Separation sequences controlled by high resolution clock in CC &S. Sequence shown in Table II
9. Command roll angle	S+3 minutes	Digital pulse train	ACS electronics	Stored command QC 1 transferred to ACS electronics
10. Command pitch angle	S+8 minutes	Digital pulse train		Inhibited unless roll angle confirmed stored command QC 2 transferred to AC electronics
11. Enable TVC subsystem	S+40 minutes	Pulse	ACS electronics	Propulsion not initiated until FC has separated 1 km from FS and roll and pitch angles have been confirmed
12. Ignite propulsion rocket	S+40 minutes	Pulse	Rocket igniters	
13. Shut off TVC electronics	S+45 minutes	Pulse	TVC	To save power
14. Command roll angle	M	Digital pulse train	ACS	Start maneuver time "M" is stored Quantitative Command (QC)5. Up to 3 maneuvers may be used. Roll angle is stored QC 6
15. Command pitch angle	M+5	Digital pulse train	ACS	Inhibited until roll angle confirmed. Pitch angle is stored QC 7
16. Arm entry pyrotechnics	E-15 minutes	Pulse	S & I device	Pyros are safed until this time
17a. Activate mach number computer	E-3 minutes	Internal signal	CC & S	See entry and descent sequences Tables XIII, XIV, and XV
17b. Deactivate attitude control. Initiate roll rate control	E-3 minutes	Pulse	ACS	Attitude control and sentry gyro shut down. Stable platform left on until impact. Roll rate controlled until parachute deployment

TABLE XI

SEPARATION SEQUENCE

Events	Time	Form	Destination	Remarks
1. Uncage platform	S-1 sec	Pulse	ACS	Separation sequence initiated by event 8c of master sequence. Sequence accomplished by 5 msec resolution clock
2. Separate FC from FS	S	Pulse	Power control	Inhibited until sterilization canister lid separation switches are open. Inhibited until separation enable command (DC 3) received
3. Enable ACS system	S+5 sec	Pulse	ACS	ACS disabled until S+5 seconds to prevent bumping. END OF SEPARATION SEQUENCE

After separation the CC&S shall transmit a digital roll angle command to the ACS electronics. After a time delay and after the ACS electronics has provided a low-level pulse to the CC&S indicating that the command angle has been achieved, the CC&S shall provide a digital pitch-angle command to the ACS electronics. The pitch and roll angle commands are stored quantitative commands (QC)1 and 2.

The thrust vector control system and the propulsion rocket shall be initiated at a fixed time after separation and after roll and pitch angles have been confirmed.

A separate timer, initiated by a separation switch shall provide a back-up rocket ignition as shown on Table XII.

At predicted time of entry, the CC&S shall be enabled to initiate pilot parachute deployment, to drop entry shell and to start the penetrometer eject sequence as a function of Mach number, altitude, or both, as described on Table XIII. At the same time the attitude control subsystem shall be switched from an attitude control to a roll rate control mode and the sentry rate control shall be deactivated. When the entry shell is deployed, the TV picture taking sequence is initiated as shown on Table XIV. In addition to controlling the TV camera shutter, the CC&S shall supply a clock pulse to the TV subsystem. The parachute and penetrometer sequence is shown inTable XV. Parachute deployment backup signal and penetrometer ejection are events which occur at predetermined times as shown in Table XIII.

6.3.2.2 Functional Block Diagram

Figure 11 shows a functional block diagram of a typical CC&S concept to accomplish the required sequential operations discussed in paragraph 6.3.2.1.

6.3.2.3 Safing and Arming

The safing and arming (S&A) subsystem shall consist of three devices.

The first device shall safe and arm all sterilization pyrotechnics, specifically the depressurization valve and canister lid pyrotechnics.

The second device shall safe and arm all pyrotechnics associated with separation, ACS and orbit injection. These are:

- Reaction control cold gas supply valves
- FS-to-FC adapter separation pyrotechnics
- Rocket igniter
- TVC propellant and ignition squibs.

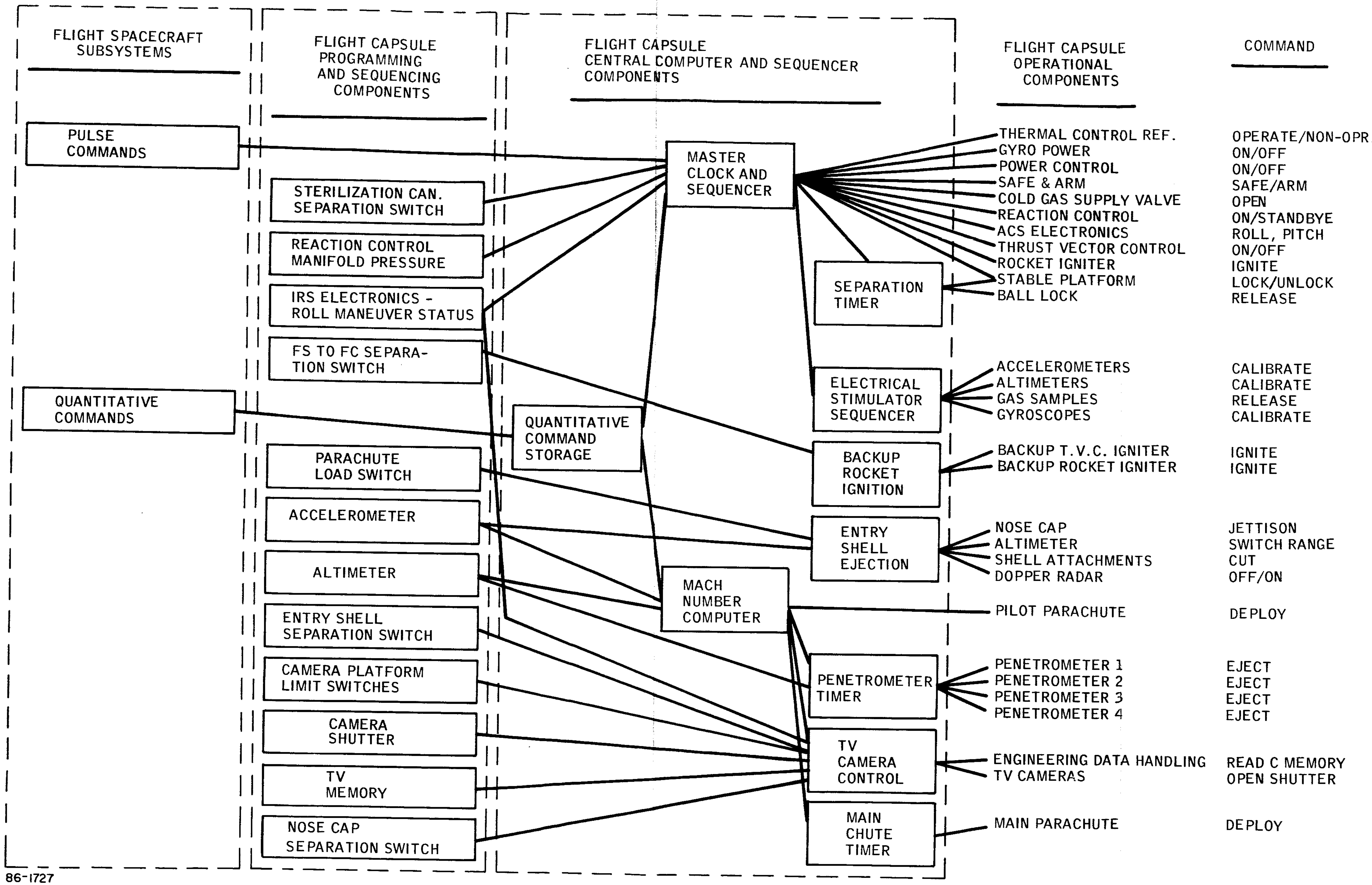

Figure 11 CC AND S FUNCTIONAL BLOCK DIAGRAM

951

TABLE XII

BACKUP DE-ORBIT SEQUENCE

Event	Time (minutes)	Input	Destination	Remarks
1. Backup retrorocket	S+43	Pulse	Power control	Backup de-orbit sequence initiated by separation switch closure and is accomplished by simple timer.
2. Backup TVC initiation	S+43	Pulse	ACS	END OF BACKUP DE-ORBIT SEQUENCE

TABLE XIII

ENTRY SEQUENCE

Event	Form	Input	Destination	Remarks
1a. Deploy pilot parachute	Pulse	Accelerometer and altimeter	Power control	Mach number computation initiated by event 17a of master sequence. Accelerometer output is square wave whose frequency is proportional to instantaneous acceleration. Accelerometer output is counted every 0.5 second. When 3 latest counts are all lower than previous count, time equals g max + 2 seconds. CC & S computes t_4, t_5 and t_6 where $t_4 = (k_1 \text{ g max})^{-A}$, $t_5 = (k_2 \text{ g max})^{-B}$ and $t_7 = (k_3 \text{ g max})^{-C}$ k_1, A, k_2, B, k_3 and C are stored QC 15, 16, 17, 18, 19 and 20. Clock resolution is 0.5 second for all entry functions. Altitudes F, G, and H are stored QC 21, QC22, and QC23. The command to deploy the pilot parachute will be given if: $t \geq t_4$ and altitude $\leq$ F or when $t \geq t_5$ or when altitude is $\leq$ G.
1b. Start TV sequence	Pulse			
1c. Start main parachute deployment sequence	Pulse			

TABLE XIII (Concl'd)

ENTRY SEQUENCE

Event	Form	Input	Destination	Remarks
2a. Cut entry shell attachments	Pulse	Load switch	Attachment pyros	Command given when load switch on parachute risers senses load or when accelerometer indicate 5g increase in deceleration in 0.5 second interval.
2b. Energize doppler radar	Pulse		Doppler radar	
2c. High altitude altimeter off	Pulse		High altitude altimeter	
2d. Low altitude altimeter on	Pulse		Low altitude altimeter	
2e. Release TV stabilization system gimbals	Pulse		Platform gimbals	
2f. Cut nose cap attachments	Pulse		Nose cap pyro	Pulse initiates 3 second pyrotechnic time delay which fires nose cap as backup to entry shell jettison
3a. Start penetrometer eject sequence	Pulse		Altimeter	Command given when altitude = H or when $t \geq t_7$
3b. Change data mode	Pulse		Accelerometer	Penetrometer data substituted for mass spectrometer data in main frame.

TABLE XIV

TV PICTURE SEQUENCE

Commands	Time (seconds)	Input	Destination	Remarks
1. Open TV camera shutter	P1	Entry shell separation switch nose cap separation switch	TV cameras	TV sequence initiated 3 seconds after entry shell or nose cap separation switch closes or 5 seconds after chute deployment signal given.
		Camera platform limit switches		Shutter command inhibited if camera platform is against stops.
		Stable platform		Shutter command unhibited if $\|\theta_c\| \leq$ 45 degrees or $\|\dot{\theta}_c\| \leq$ 13 degrees second. This is a backup in the event that the camera platform fails.
				The above inhibit functions over ridden 8.2 seconds after TV sequence initiated.
2. Start reading C memory	P1+6		Eng. data handling	EDH must start reading data out of memory 6 seconds after P1.
		Shutter confirmation pulse		Shutter command inhibited for 12 seconds after shutter confirmation pulse received.
3. Open TV camera shutter	P2	Memory status pulse		TV sequence reinitiated when all but 14.2 seconds of data has been played out of memory.
		Camera platform limit switches		Camera platform is against stops, shutter command inhibit function over ridden 8.2 seconds after TV sequence reinitiated.
				Shutter command inhibited for 12 seconds after shutter confirmation pulse received.
				Event 3 is repeated each time all but 14.2 seconds of data has been played out of memory.

TABLE XV

PARACHUTE AND PENETROMETER SEQUENCE

Event	Time (seconds)	Form	Destination	Remarks
1. Deploy main parachute	Pilot chute Deploy +2 = t_6	Pulse	Power control	Backup to pilot parachute deployment. Gas generator forces main parachute out of container 2 seconds after pilot parachute deployment signal is generated.
2. Backup nose cap jettison	t_6+3			
3. Eject first penetrometer	t_7		Penetrometer ejector	Penetrometer ejection sequence initiated as function of altitude H or time t_7 from maximum g
4. Eject second penetrometer	t_7+2		Penetrometer ejector	
5. Eject third penetrometer	t_7+4		Penetrometer ejector	
6. Eject fourth penetrometer	t_7 +6	Pulse	Penetrometer ejector	

The third device shall safe and arm all pyrotechnics associated with entry. These include:

- Nose cap ejection squibs
- Pilot parachute deployment squibs
- Main parachute deployment squibs
- Entry shell attachment squibs
- Penetrometer eject squibs

1. <u>Operating and Design Requirements</u>

   a. Each device shall contain a series of two sets of redundant switches which shall isolate each pyrotechnic from undesired signals. The first pair of switches shall be inertially actuated and the second set shall be command actuated. The series of switches shall be interconnected such that events such as sustained acceleration and an electrical command must occur in proper sequence for the S&A device to arm. The interconnections shall also provide that if either one of the redundant switches fails to arm, the S&A device shall be armed by the remaining switch of that pair.

   b. An inertial switch shall close the circuit when it has been subjected to an acceleration of 5 g or greater for a period of 10 seconds. Once closed, the inertial switch shall remain closed unless remotely reset.

   c. An electrically operated switch with two coils shall close the circuit in response to a pulse on one coil and remain closed unless a pulse of the same polarity is applied to the second coil. The electrical switch shall provide a short circuit across the pyrotechnic when in the safe position.

   d. A separate set of monitoring contacts shall be provided in each switch. The monitor contacts shall be interconnected in such a manner that continuity shall exist between two terminals when all four switches are in the safe (circuit open) position but that continuity shall be broken if any of the four switches are in the armed (circuit closed) position.

e. Each S&A device shall include an integral electrical interconnecting cable to each set of pyrotechnics which has no breaks between the S&A device and the connector which attaches directly to the pyrotechnic.

f. Each armed circuit shall be capable of carrying 4.5 amps at 28 + 4, -10 vdc.

g. The electrical resistance between the input and output connectors of the armed system shall not exceed 1 ohm.

h. Isolation between circuits shall be 1 megohm.

i. The electrical switch shall operate on receipt of a 1 amp, 28 vdc, 0.4 second pulse.

j. The rocket igniters, and the squibs used to detonate the flexible mild-detonating explosive in the sterile canister lid constitute hazards to ground and test personnel. These pyrotechnic initiators shall be physically isolated from the propellant or explosive by a pressure sensitive device whenever the ambient pressure is greater than 3.7 psia and shall be in physical contact with the propellant or explosive whenever the ambient pressure is less than 1.0 psia.

k. In addition to the above requirements for the first two S & A devices, the third device, which is associated with entry pyrotechnics, shall contain a third set of switches. The first pair shall be inertially actuated and the third pair shall be command actuated. The interconnections shall provide for adherence to proper sequence and for redundant backup, as before. The pressure switch shall open the circuit at pressures of 3.7 psia or greater and close the circuit at 1.0 psia or less.

6.3.3. Functional Interfaces

6.3.3.1 Electrical Inputs

1. Discrete commands (DC) from the Flight Spacecraft to the programming and sequencing components:

DC 1	Start master sequence
DC 2	Inhibit override - stable platform drift
DC 3	Separation enable
DC 4	Spare
DC 5	Spare
DC 6	Start checkout sequence
DC 7	Release calibration gas sample
DC 8	Arm separation system
DC 9	Start electrical stimulation sequence
DC10	Open cold gas supply valve
DC11	Separate FC from FS
DC12	Inhibit override - sterile canister lid separation switch
DC13	Inhibit override - reaction control pressure

2. Quantitative commands (QC) from the Flight Spacecraft to the CC & S quantitative command storage:

QC 1	Thrust vector roll command
QC 2	Thrust vector pitch command
QC 3	Spare
QC 4	Spare
QC 5	Time of first cruise maneuver sequence

QC 6	First maneuver roll angle
QC 7	First maneuver pitch angle
QC 8	Time of second maneuver sequence
QC 9	Second maneuver roll angle
QC10	Second maneuver pitch angle
QC11	Time of third maneuver sequence
QC12	Third maneuver roll angle
QC13	Third maneuver pitch angle
QC14	Predicted time of entry
QC15	Constant K_1 for maximum parachute deployment Mach number
QC16	Constant A for maximum parachute deployment Mach number
QC17	Constant K_2 for minimum parachute deployment Mach number
QC18	Constant B for minimum parachute deployment Mach number
QC19	Constant K_3 for penetrometer deployment Mach number
QC20	Constant C for penetrometer deployment Mach number
QC21	Altitude F maximum parachute deployment altitude
QC22	Altitude G minimum parachute deployment altitude
QC23	Altitude H penetrometer deployment altitude

3. Signals from various Flight Capsule instrumentation to the CC&S components:

Signals	Form of Signal
1. FC-to-FS separation	open/closed
2. Sterilization canister lid separation	open/closed
3. Acceleration	square wave-frequency proportional to acceleration
4. Parachute riser line load	open/closed
5. Pressure - sterilization canister	on/off
6. Nose cap separation	on/off
7. Altitude - above 27, 500 feet	digital pulse train
8. Altitude - below 27, 500 feet	digital pulse train
9. Camera platform, limit switch	on/off
10. Altitude confirmation	on/off

4. Power -- As listed in Table XVI.

6.3.3.2 Outputs -- As listed in Table XVI.

6.3.3.3 Thermal

Electrical power, insulation, and surface coatings as required to maintain the temperature within the limits shown on Table XVI.

6.3.4 <u>Component List</u>

6.3.4.1 Programming and Sequencing Subsystem

1. Pressure switch--reaction control manifold
2. Separation switch--sterilization canister lid
3. Separation switch--FS to FC separation
4. Separation switch--FS to FC separation
5. Load switch--parachute

TABLE XVI

FUNCTIONAL INTERFACES

Function	CC&S	Pressure Switches	Separation Switches	Load Switch	Accelerometer	Altimeter	Stable Platform	Sterile Canister S&A	Separation & Entry S&A	Isolator Squib	Isolator Igniter
Inputs	6 Watts 28 ± 1 vdc	28 ± 1 vdc	28 ± 1 vdc	28 ± 1 vdc	See par. 6.9.2	See par. 6.9.13	See par. 6.5	From FS 28 V pulse to arm 28 V pulse to safe 4.5 Amp pulse to fire	From CC&S 28 V pulse to arm 28 V pulse to safe 4.5 Amp pulse to fire		
Outputs	See Tables VIII thru XV clock frequency between 1 and 100 kilocycles	0/28 vdc	0/28 vdc	0/28 vdc				4.5 Amp pulse to squibs	4.5 Amp pulse to squibs		
Weight	8 pounds	0.1 pound	0.1 pound	0.1 pound				1.5 pounds	1.5 pounds	0.2 pound	0.2 pound
Volume	300 in.^{3}	4 in.^{3}	4 in.^{3}	4 in.^{3}				36 in.^{3}	36 in.^{3}	5 in.^{3}	5 in.^{3}
Operating temperature limits (1)	0 to +175°F	-65 to +175°F	-65 to 175°F	-65 to 175°F				0 to 175°F	0 to 175°F	0 to 175°F	0 to 175°F
Nonoperating temperature limits	-65 to 175°F	-65 to 175°F	-65 to 175°F	-65 to 175°F				-65 to 175°F	-65 to 175°F	-65 to 175°F	-65 to 175°F

(1) Exclusive of sterilization

106-2

6. Accelerometer--X-axis

7. Altimeter--high altitude

8. Altimeter--low altitude

9. Limit switch--camera platform

10. Electronic package--stable platform (for altitude confirmation)

11. CC&S

6.3.4.2 Safing and Arming Subsystem

1. S & A device--entry pyrotechnics

2. S & A device-- sterilization canister and separation

3. Isolater--squib

4. Isolater--igniter

6.4 DE-ORBIT

6.4.1 Function

The de-orbit function shall provide a propulstion subsystem capable of decreasing the velocity of the Separated Vehicle to place it on an impact trajectory.

6.4.2 Performance/Design Requirements

6.4.2.1 Performance

1. Total Impulse, lb-sec -- 101, 600 ± 1 percent, 3 sigma

2. Specific Impulse, second (Voltsac)--> 254

3. Thrust Nominal, pounds -- 3000

4. Temperature Environment -- exclusive of sterilization:

   a. Storage -40 to 175°F

   b. Operation -40 to 175°F

5. Space Storage -- up to one year at 10^{-6} mm Hg.

6. Operating Environments -- see paragraph 4.3

6.4.2.2 Design

1. Propellant Mass Fraction

   <0.90

2. Type Rocket Motor

   Solid propellant spherical rocket motor with submerged nozzle. The unit shall be sterilizable.

3. Allowable Envelope

   a. Diameter, maximum -- 24 inches

   b. Length, Maximum -- 25 inches

   c. Ignitor Location -- installed at nozzle end of motor case

4. Engine Exhaust Products

   Gaseous only

5. Thrust Vector Alignment

   The alingment shall be within a 0.5 degree cone angle around the theoretical thrust axis

6. Note

   Existing qualified rocket components shall be used where applicable.

6.4.3 Functional Interfaces

6.4.3.1 Electrical

1. Inputs -- 4.5 amperes ignition pulse

2. Outputs -- none

6.4.3.2 Mechanical

1. Mounting Installation -- Provide mounting flange with bolt hole circle of 23.25 inches diameter. Flange plane shall be perpendicular to motor thrust axis located 14.5 inches from the motor head end, and within the alignment and accuracy requirements in paragraph 4.4.

2. Exhaust Nozzle Extension -- Provide mounting for an exhaust nozzle extension to be connected to the motor nozzle exit plane within the alignment and accuracy requirements in paragraph 4.4.

6.4.4 Component List

6.4.4.1 Rocket Motor

6.4.4.2 Exhaust Nozzle Extension

6.5 ATTITUDE CONTROL

6.5.1 Functions

The attitude control functional area, hereafter called the attitude control subsystem (ACS), shall be capable of performing the following functions:

6.5.1.1 Nullify the separation rates and realign the Separated Vehicle to the Flight Spacecraft reference attitude.

6.5.1.2 Orient the Separated Vehicle to the correct attitude for thrust application to de-orbit.

6.5.1.3 Maintain the Separated Vehicle attitude during the thrust application phase.

6.5.1.4 Orient the Entry Vehicle to a preferred attitude during the cruise mode to optimize communications and entry conditions.

6.5.1.5 Maintain roll control through entry up to parachute deployment

6.5.1.6 Provide three-axis accelerometer data for controlling entry and descent events and to provide information for analyzing atmospheric characteristics including wind profiles.

6.5.1.7 Provide control signals to maintain the two-axis television camera gimbal system aligned to the local vertical.

6.5.2 Performance/Design Requirements

6.5.2.1 Attitude Control Subsystem (ACS) Operation

An attitude reference shall be established on the Separated Vehicle by aligning a four-gimbal inertial platform with respect to the Flight Spacecraft attitude reference (Sun-Canopus). The required reorientation angles shall be stored in the CC & S memory. At separation, the Separated Vehicle platform shall be operating in an inertial mode and the Separated Vehicle shall then be commanded, via the stored angle command, to the required thrust application attitude. The attitude control subsystem shall maintain this attitude by means of cold gas reaction control subsystem until thrust application. Just prior to thrust application, a hot gas reaction subsystem (yaw and pitch) shall be activated to provide stabilization during thrust application. The cold gas subsystem shall remain active during the thrusting phase and the cold gas roll nozzles shall provide roll stabilization during this phase as well as through entry and up to parachute deployment.

On completion of the thrusting phase, the attitude control subsystem shall maintain the attitude of the Entry Vehicle using the cold gas reaction subsystem. The ACS shall, of so commanded, reorient and hold the vehicle to a preferred attitude for communication during the cruise mode. It shall orient the vehicle to a preferred attitude prior to entry to minimize the angle of attack. These angular commands will be stored in the CC & S and be issued to the inertial reference subsystem (IRS) computer at the correct time.

The ACS shall contain a subsystem that will disable the reaction control subsystem during the cruise mode should the angular rate exceed 6 deg/sec about an axis, to limit the tumble rate at entry in case of an ACS failure.

During entry, the reaction control subsystem shall continue to provide roll rate control. In addition, the inertial reference subsystem instrumentation shall provide acceleration data during the entry phase as engineering information and for event control purposes. On parachute deployment, the IRS shall provide the TV camera gimbal subsystem with the required commands to maintain the optical axis of the cameras along the local vertical.

6.5.2.2 Attitude Control Subsystem Nominal Performance Summary

1. The overall ACS performance is characterized by:

   a. Performance -- during thrusting phase, 0.5 degrees (1σ), at entry, 1.0 degree (1σ).

b. Operating Times -- Separation to thrusting, 0.5 hour (max.); thrusting phase, 35 seconds (max.); Cruise phase, 1.0 hour (max.).

c. Weight -- 90 pounds (max.).

d. Stored impulse -- Hot gas, 3500 lb-sec (max.) Cold gas, 248 lb-sec (max.).

e. Limit Cycle Amplitude -- Yaw and Pitch, 0.5 degree (max.); Roll, 1.0 degree (max.)

2. The IRS performance for use after entry is characterized by:

a. Pointing Performance -- 1.0 degree (1 σ).

b. Operating Time -- 0.5 hour.

6.5.2.3 Inertial Reference Subsystem (IRS) Operation

The inertial reference subsystem shall consist of a four-gimbal inertial platform, a digital computer and a 3-axis rate sensor for rate limiting. The four gimbals on the platform are required to permit full flexibility and angular freedom. The platform shall also contain the accelerometers used for both control functions and engineering data, thus providing inertially referenced acceleration data. The digital computer shall provide the means of transforming the platform gimbal angles into the proper reference frame for commanding the vehicle reaction control subsystem. The computer shall accept angular commands from the CC & S and provide the logic to control the reaction control valves. The computer shall perform the proper transformations from the IRS gimbal angles to command the TV camera gimbal angles along the local vertical.

6.5.2.4 Inertial Reference Subsystem Performance/Design Summary

1. Control Logic -- The control "deadband" shall be ± 0.5 degrees from the nominal commanded angle for the yaw and pitch axis and ± 1.0 degrees for the roll axis with a hysteresis factor of 10 percent.

2. Error Sources -- Error contributions (1 sigma) in the computation and control of the angles are:

G-insensitive gyro drift	0.4 deg/hr
G-sensitive gyro drift	0.3 deg/hr

Gimbal readout error	0.1 degree
Computation error	0.1 degree

3. Weight and Volume Summary

Component	Weight (pounds)	Volume (cubic inches)	Power (watts)
Inertial platform	10	400	45
Computer	5	160	10
Rate gyro limiter	2	50	13
Cables	1	---	---
Totals	18	610	68

6.5.2.5 Cold Gas Reaction Subsystem Operation

The reaction control subsystem for the nullification of separation rates, orientation, roll control during de-orbit thrusting, reorientation and limit cycling shall utilize gaseous nitrogen as the propellant. This cold gas subsystem shall consist of two identical subsystems each utilizing a central regulator, pressure vessel, vent and fill manifold, squib valve, filters, and solenoid valve nozzle assemblies. The twelve solenoid valve nozzle packs provide the required three-axis control in couples. This redundant approach shall be utilized to provide the necessary mission safety margin in the event of failure modes such as:

1. Developing a gas leak in one of the subsystems

2. A valve failing to close.

In the case of one of these failures, one nozzle of the couple will be inoperative but the other nozzle will allow completion of the mission.

6.5.2.6 Cold Gas Reaction Components Performance/Design Summary

1. Thrust Levels per Axis

   a. Yaw - 1.0 pounds (1 lb at 7.33 ft = 7.33 ft-lb.)

   b. Pitch - 1.0 pound (1 lb at 7.33 ft = 7.33 ft-lb)

   c. Roll - 1.0 pound (1 lb at 7.33 ft = 7.33 ft-lb)

2. <u>N_2 Total Impulse</u> --

   a. Required - 75 lb-sec.

   b. Stored - 248 lb-sec.

3. <u>Response Parameters</u> --

   a. Time delay - 0.020 second

   b. Time constant - 0.005 second

4. <u>Mechanical Properties</u>

Component	Weight each (pounds)	Number	Weight (pounds)
Solid nozzle valve	0.18	12	2.2
Pressure vessel	5.90	2	11.8
Gas (N_2)	5.00	--	5.0
Squib Valves	0.37	2	0.7
Vessel manifold	0.25	2	0.5
Fill manifold	0.75	2	1.5
Tubing	2.85	2	5.7
Regulator	3.00	2	6.0
Filter	0.25	4	1.0
		Total	34.4

6.5.2.7 Solid Propellant Hot Gas Reaction Subsystem Operation

A hot gas reaction subsystem shall maintain the desired vehicle attitude during the de-orbit thrust application. This system shall be composed of four identical solid propellant gas generator packs each consisting of a solid propellant gas generator, closely coupled to two solenoid operated hot gas valves and two reaction nozzles. These packs shall be located on the vehicle as couples on both the pitch and yaw axis, to increase mission safety margin in case of a generator or valve failure.

6.5.2.8 Solid Propellant Hot Gas Reaction Component Performance/ Design Summary

1. Thrust Levels per Axis

   a. Yaw -- 25 pounds each of two nozzles (50 lb total at 7.33 ft = 366.5 ft-lb)

   b. Pitch -- 25 Pounds each of two nozzles (50 lb total at 7.33 ft = 366.5 ft-lb)

2. Gas Generator Performance

   a. Required total impulse - 1250 lb-sec. (per axis)

   b. Stored total impulse - 3500 lb-sec. (all axis)

   c. Specific impulse - 180 seconds

   d. Burn time - 35 seconds

3. Response Parameters

   a. Time delay 0.020 seconds

   b. Time constant 0.005 second

4. Weight Summary

Component	Weight each (pounds)	Number	Weight (pounds)
Gas generator	7.77	4	31.1
Solenoid nozzle valve	0.52	8	4.2
Tubing brackets, etc.			5.0
			Total 40.3

5. Temperature Limits

   a. Operation -100 to 140°F

   b. Storage -100 to 140°F

   c. Sterilization 300°F maximum

6.5.3 Functional Interfaces

6.5.3.1 Electrical

1. Inputs -- The IRS computer shall receive and store the commanded angles for the CC & S. The CC & S shall also provide a number of discrete commands to initiate various sequences in the ACS (activate cold gas reaction subsystem, activate hot gas subsystem, etc.). The attitude control subsystem shall receive electrical power from the electrical power and control subsystem.

2. Outputs -- The IRS shall provide command signals to the TV camera gimbal during the parachute descent phase and accelerometer data to the CC & S and telemetry subsystem signals to provide diagnostic measurements to monitor subsystem performance.

6.5.3.2 Mechanical

1. Cold Gas Reaction Subsystem -- The cold gas reaction control subsystem shall produce torques in couples about all axes. The nozzles shall be located at the outer diameter of the vehicle, thus providing an 88-inch moment arm.

2. Hot Gas Reaction Subsystem -- A hot gas subsystem shall provide control of pitch and yaw disturbing torques during thrusting for the rocket engine. The hot gas nozzles shall also have an 88-inch moment arm.

3. Deorbit Rocket Engine -- Allowable 3 sigma errors for the rocket engine shall not exceed:

   a. Thrust vector misalignment = 0.5 degree

   b. Rocket location error = 0.06 inch.

4. Separated Vehicle Center of Gravity -- The Separated Vehicle center of gravity location error shall not exceed 0.5 inch (3 sigma). The initial attitude error at separation from the Flight Spacecraft shall not exceed 0.7 degree (3 sigma). This error includes the contributions from Flight Spacecraft attitude uncertainty, limit cycle, and alignment of the IRS in the Flight Spacecraft.

6.5.4 Components List

The ACS consists of the following major subsystems:

1. Inertial reference subsystem

2. Cold gas reaction subsystem

3. Hot-gas TVC subsystem

6.6 DESCENT RETARDATION - ENTRY

6.6.1 Functions

The functions of descent retardation - entry, hereafter termed the entry shell, are divided into two categories; 1) those functions associated with the primary and afterbody structure which provides for loads, shape and associated subsystem attachment, and 2) those functions associated with primary and secondary heat shield.

The primary structure shall provide the vehicle shape for atmospheric deceleration, mounting and hardpoint for the Flight Capsule subsystems, and surfaces for thermal protection which include the primary and secondary heat shields. The afterbody structure, in addition to providing surfaces for part of the secondary heat shield, shall provide a shape of sufficient aerodynamic instability to impart a pitching moment to reorient the vehicle to the blunt end forward position in the event of rearward entry.

The primary heat shield shall provide the thermal protection required to prevent excessive heating of the primary structure and Entry Vehicle interior during entry. The secondary heat shield (that portion of heat shield over the afterbody and exposed interior portion of the primary structure) shall provide thermal protection to the aft portion of the Entry Vehicle during the normal blunted end forward entry mode. Additionally, it shall provide thermal protection in the event of rearward entry.

6.6.2 Performance/Design Requirements

6.6.2.1 Preentry

1. Structure -- The entry shell structure shall be subjected to the environments described in Figure 4. The entry performance of the shell shall not be degraded by these environments, particularly the sterilization heating cycle, launch dynamic environment and space flight temperature distributions. The entry structure shall be compatible with the heat shield to prevent cracking of the ablator, failure of the heat shield-to-structure bond or excessive stress in the structure induced by relative thermal expansion over the expected temperature ranges.

2. Heat shield -- The heat shield (with the thermal control coating applied to it) shall be subjected to the environment shown in Figure 4. The thermal protection capability of the heat shield shall not be degraded by exposure to these environments. The heat shield shall be compatible with both the thermal control coating and with the underlying structure to prevent degradation of their functions or its own fuctions during entry phase.

6.6.2.2 Entry

1. Structure -- The environment that the entry shell structure shall withstand during entry is given in Figure 4. The shell shall be stable under all pressure distributions including unsymmetrical distributions induced during entry. No permanent unsymmetrical deformations are permissible. The geometry of the outer surface of the entry shell structure shall consist of a spherical cap having a nominal radius of curvature of 22.5 inches, joined to a truncated cone with a continuously turning tangent. The semivertex angle of the truncated cone shall be 60 degrees. The nominal base radius shall be 90 inches. The conical shell construction shall be aluminum honeycomb sandwich. Suitable stiffening shall be made at the outer edge to prevent shell collapse.

2. Heat Shield

a. The heat shield shall withstand the entry environments and perform its function under all of the critical conditions including the specified failure modes.

b. The heat shield shall be applied on an aerodynamic shape with an hypersonic drag coefficient of 1.63 ± 0.08. This configuration shall be a spherically blunted cone ($R_N/R_B = 0.25$, $\beta = 60$ degrees) which defines the structural outer shell contour. The maximum shoulder radius, with heat shield applied, shall not exceed 0.01D. Maximum variation in heat shield thickness (from minimum to maximum on conic) shall be 2 inches.

c. The heat shield shall maintain its integrity through the entry phase to parachute opening under heating and aerodynamic loads and shall operate according to an ablation-heat absorption-rejection principle. It shall be made from Purple Blend, Mod 5, or similar ablator, bonded by suitable means to the shell structure. The maximum bond line temperature shall be consistent with the shell structure design requirements. Temperature tolerance on the bonding materials shall be consistent with the shear and normal forces it must transmit (500 ± 100°F). The initial temperature at entry (800,000 feet) shall be consistent with the thermal control requirements and capability within the range of 50 to 150°F, to assure light weight heat shield design. Deviations from these temperature limitations are permissible if warranted by the overall system weight tradeoff studies.

6.6.3 Functional Interfaces

6.6.3.1 Thermal Control

The entry shell structure and entry thermal protection interface occurs at the bond line between the two systems. The bond shall transmit shear and normal forces induced by relative thermal expansion and external surface pressure.

The entry shell shall provide a support for the thermal control coating and also provide a part of the conduction path for the heat flow.

6.6.3.2 Attitude Control

The entry shell shall provide support and transmit control torques of the reaction control and thrust vector control nozzles.

6.6.4 Component List

6.6.4.1 Primary Structure

6.6.4.2 Primary Heat Shield

1. Heat absorbing material (ablator)
2. Bond
3. Substructure

6.6.4.3 Secondary Heat Shield

1. Ablator
2. Bond
3. Substructure

6.6.4.4 Afterbody Heat Shield

1. Ablator
2. Bond
3. Substructure

6.6.4.5 Miscellaneous Subsystem Protective Shields (antennas, TV, etc.).

6.7 DESCENT RETARDATION - POSTENTRY

6.7.1 Functions

6.7.1.1 Primary

The primary function of descent retardation - postentry, hereafter called the parachute descent subsystem, shall be to decelerate the

Suspended Capsule to a velocity which allows a preselected parachute descent time from a minimum for data playout to a maximum compatible with Flight Spacecraft orbital constraints and scientific data acquisition.

6.7.1.2 Secondary

A secondary function of the parachute descent subsystem is to separate the Suspended Capsule from the entry shell.

6.7.2 Performance/Design Requirements

6.7.2.1 Parachute Descent Subsystem Operation

1. Descent Time -- The subsystem shall be designed to sense, analyze, and utilize entry parameters (peak acceleration and time), to initiate deployment of a main parachute which will decelerate the Suspended Capsule, to provide the required minimum descent time of 160 seconds for data playout, but not to exceed 360 seconds to be compatible with scientific data acquisition, yet remain within the constraints of the Flight Spacecraft relay-link geometry.

2. Entry Shell -- The subsystem shall decelerate the Suspended Capsule to separate it from the entry shell.

3. Environment -- The subsystem shall perform its intended function while exposed to the general environments specified in Figure 4 and the specific environments specified below.

a. Prelaunch -- Equivalent to 2 years of shelf life plus transportation.

b. Sterilization -- Three cycles of 36 hours at 145°F in a nitrogen atmosphere plus exposure to a chemical atmosphere of 12 percent ethylene oxide and 88 percent Freon 12 at a relative humidity of 50 percent for a 5-day period of 140°F.

c. Space -- Duration: 300 days;
Vacuum: 1×10^{-8} torr
Radiation: 6×10^{4} rads
Temperature: -85 to + 120°F.

d. Operation -- Duration: 60 minutes
Atmosphere: Carbon Dioxide
Temperature: -100 to + 150°F
Shock: 200g

4. Deployment Conditions -- The subsystem shall be deployed at 27,500 feet ($M < 1.2$) in the VM-3, 4, and 7 atmospheres and at Mach 1.2 ($Z \simeq 21,000$ feet) in the VM-8 atmosphere.

5. Material -- The parachute canopy fabric, riser line, suspension lines and harness attachment assembly shall be designed to withstand the snatch force and opening shock load for the worst design conditions.

6. Entry Angle and Atmosphere -- The parachute descent subsystem shall accommodate the design combinations of entry angle and atmosphere to be compatible with the deployment Mach number ≤ 1.2 and the deployment altitude $\leq 27,500$ feet.

6.7.2.2 Subsystem Component Requirements

The components of the parachute descent subsystem shall operate as outlined within the limits specified.

1. Deployment Initiation Sensing and Sequence Control

a. Sensing -- The sensing subsystem shall consist of a three-axis accelerometer to sense peak acceleration, a timer used to measure time from peak g to pilot parachute deployment, an analog electrical device relating peak g and time, and a radar altimeter. The subsystem shall sense and monitor the vehicle acceleration along the flight path within an accuracy of 1.0 percent over a range of accelerations from 3 to 25 earth g. The subsystem shall retain the maximum acceleration level and submit this level to the sequence control subsystem.

b. Sequence Control -- The sequence control subsystem shall accept the maximum acceleration signal from the sensing subsystem and determine the proper time to initiate the deployment pyro firing signal. The time measurement error shall be less than 3 percent for time delays from peak acceleration to deployment, ranging between 40 and 175 seconds for Mach 1.2 correlation. The radar altimeter shall be operated in series with the g peak, time correlation to limit deployment to 27,500 feet or lower.

c. Radar Altimeter -- A radar altimeter capable of measuring true altitude above the surface within an accuracy of 2.0 percent over a range of 500 to 40,000 feet shall be used to perform the following deployment operations. During entry the altitude shall be continually measured to sense 27,500 feet. If at this time the vehicle has decelerated to Mach 1.2 or less, deployment shall take place (VM-3, 4 and 7 atmospheres). If Mach 1.2 cannot be achieved at 27,500

(VM-8 atmosphere) the Suspended Capsule shall decelerate to Mach 1.2 at some lower altitude (typically 21,000 feet) at which time deployment shall take place.

d. Deployment Backup Modes -- If the Suspended Capsule reaches an altitude of 20,000 feet and no deployment signal has been received, an override firing signal to the pilot parachute system shall be transmitted as a backup deployment mode due to an accelerometer malfunction. In the event of radar altimeter malfunction, a peak g and time correlation based on Mach 0.85 shall be utilized.

e. Pilot Parachute Deployment -- On receipt of the firing signal, the pilot parachute mortar shall fire, breaking the pressure seal and deploying the pilot parachute. The mortar shall provide 100 ft/sec. of separation velocity over a 10g axial acceleration level.

2. Pilot Parachute -- The pilot parachute shall remove the main parachute canister cover and extract the main parachute in its deployment bag. At main parachute line stretch, the pilot parachute shall strip the deployment bag from the main parachute to allow it to fully inflate.

3. Main Parachute

a. Storage canister -- A storage canister shall provide the main parachute with the required environment.

b. Deployment bag -- A deployment bag shall contain the main parachute during extraction to deployment position.

c. Main parachute -- The main parachute shall provide the minimum and maximum required descent time.

d. Mortar assembly -- A mortar assembly shall be designed to eject the pilot parachute at a signal from the sensing system.

e. Gas generator -- In the event the mortar assembly fails to eject the pilot parachute, a gas generator shall be provided as a backup to deploy the entire main parachute assembly. One second after indication of the mortar malfunction a backup signal shall be provided to initiate the gas generator to deploy the main parachute.

6.7.3 Functional Interfaces

6.7.3.1 Electrical

1. Outputs from Tension Switch -- The tension switch to senses parachute snatch loads and signals the CC&S for appropriate sequencing.

2. Outputs from Pyrotechnic Continuity Loops -- Two for mortar initiation igniters, and two for gas generator igniters.

3. Inputs from CC&S

a. Receive signal from CC&S to ignite mortar propellant to fire mortar sabot which ejects the pilot parachute as the primary mode for main parachute deployment.

b. Receive signal from CC&S to ignite the gas generator propellant which fills the gas generator bag, which in turn ejects the main parachute as a backup mode of deployment.

6.7.3.2 Mechanical

1. Main Parachute Canister -- Structural mounting to the Suspended Capsule structure.

2. Main Parachute Harness -- Attachment to the Suspended Capsule structure.

6.7.3.3 Thermal

Passive thermal control means shall be utilized to maintain the component temperatures within the desired range during all environments.

6.7.4 Component List

1. Main parachute (ring-sail type)
2. Swivel
3. Harness assembly and attachments
4. Main parachute deployment bag
5. Main parachute canister
6. Gas generator bag
7. Gas generator propellant container
8. Pilot parachute (ring-slot type)
9. Pilot parachute mortar assembly.

6.8 THERMAL CONTROL

6.8.1 Function

The function of thermal control shall be to maintain the Flight Capsule energy balance within specified temperature limitations while subjected to widely varying thermal environments during the following distinct operational modes:

1. Sterilization
2. Factory to launch
3. Launch
4. Cruise
5. Midcourse maneuver near Earth
6. Maneuver near Mars
7. Planetary orbit
8. Flight Spacecraft separation
9. Postseparation Separated Vehicle cruise
10. Planetary entry
11. Parachute descent.

6.8.2 Performance/Design Requirements

The thermal control concept shall be passive, supported by heat addition to critical areas where required. Temperature balancing shall primarily be achieved by controlled radiative energy interchange between components, controlled exchange between components and space as well as controlled heat leakage. Radiation and heat leakage shall be controlled by surface preparation or the application of coatings with specified radiative properties to surfaces, thermal insulation and the specification of contact resistances between joints where applicable.

6.8.2.1 Sterilization Canister

1. Canister Lid

a. Surface -- the emissivity of the external and internal surface shall be low to minimize heat losses to space during cruise.

2. Canister base

a. Surface -- The emissivity of the external surface shall be low to reduce heat losses to space. The emissivity of the internal surface shall also be low to minimize disturbances to the thermal balance of the Flight Spacecraft after separation of the Flight Capsule.

6.8.2.2 Entry Shell

Temperature limitations, minimum and maximum operating temperature levels shall be determined to assure heat shield/structure compatibility during the mission sequence and survival through the sterilization process without degradation of the heat shield material, consistent with a minimum heat shield weight for atmospheric entry heat protection.

1. Entry Structure Surface -- The temperature range of the heat shield/structure bond and heat shield material requirements. A thermostatically controlled redundant heating grid shall be located in the entry shell for heat shield thermal control during cruise and planetary orbit.

2. Primary Heat Shield -- The emissivity of the surface shall be low to minimize heat losses after sterilization canister lid ejection.

3. Secondary Heat Shield -- The ratio of absorbtivity to emissivity, α/ϵ, governs the Entry Vehicle postseparation temperatures and actual secondary heat shield surface requirements related to orbital parameters. The α/ϵ ratio shall be selected to be compatible with these parameters to minimize entry heat shield temperatures.

6.8.2.3 Afterbody

1. Temperature Limitations -- Same requirements as paragraph 6.8.2.2, Secondary Heat Shield.

2. Support Structure Surface -- No particular requirements.

6.8.2.4 Subsystem Modules

1. Temperature Limitations

a. Telecommunication

Long-time, nonoperation:	-60 to +275°F
Operating:	0 to +175°F
Sterilization cycle:	+300°F

b. Instrumentation

Long-time nonoperating:	-40 to +140° F
Operating:	0 to +100° F
Sterilization cycle:	+300° F

2. Surfaces -- The emissivity of the subsystem modules external surfaces shall be low to minimize radiative heat exchange with the surrounding environment. The emissivity of the internal surfaces shall be high to promote heat transfer by radiation from heat generative components and to fully utilize the heat sink capability of the subsystem module housing. The emissivity of component surfaces shall maintain the energy balance within the modules within specified temperature limitations during all mission phases. High emitting surfaces shall be utilized where heat dissipation is required, low emitting surfaces where heat dissipation is undesirable.

3. Insulation

a. The subsystem module housings shall be thermally insulated from the structure to reduce transient temperature variations which affect the module thermal balance.

b. Efficient thermal conduction paths shall be provided between the individual components to fully utilize the subsystem module housing heat sink capability.

c. Superinsulation blankets shall be provided between the subsystem modules and the payload compartment as required to reduce environmental heat exchange.

4. Heat -- A thermostatically controlled redundant system of heaters shall be located in each module to compensate for environmental variations during the various mission phases.

6.8.2.5 Rocket Engine

1. Temperature Limitations

Long-time, nonoperations:	-40 to +160° F
Before operation:	-40 to +175° F
Sterilization cycle:	+300° F

2. Surface -- Surfaces exposed to space after Flight Spacecraft separation shall have an α/ϵ ratio similar to the afterbody as discussed in paragraph 6.8.2.3. All other surfaces have no particular requirement.

3. Heat -- A redundant heating blanket system shall be located around the rocket case if the long-time nonoperating temperatures are below the limits in 1. above. Each system shall have a thermostat which controls rocket engine temperature limitations.

6.8.2.6 ACS Reaction Subsystem

1. Temperature Limitations

Operating and nonoperating: -100 to +275°F
Sterilization cycle: +300°F

2. Surface -- Temperature limits of the ACS reaction subsystem are expected to be within the same limits as the entry shell temperatures. The α/ϵ ratio for the ACS reaction subsystem shall be selected in accordance with paragraph 6.8.2.2, Secondary Heat Shield.

3. Insulation -- An efficient conduction path to the entry shell shall be provided to maximize conductive heat exchange.

6.8.2.7 Separation Subsystems

Thermal insulation shall be provided for each separation subsystem as required. Separation subsystems to be considered include the following:

1. Flight Capsule separation
2. Sterilization canister lid separation
3. Separated Vehicle separation
4. Nose cap separation
5. Parachute ejection

6. Entry shell separation

7. Penetrometer separation

6.8.3 Functional Interfaces

6.8.3.1 The heat path between the canister base and entry structure shall be designed to maximize heat transfer by conduction.

6.8.3.2 The heat path between the canister base and canister lid shall be designed to maximize heat transfer by conduction.

6.8.3.3 The thermal resistance of the heating pads, located between the primary heat shield and the canister lid, shall be maximized to minimize heat losses to space during cruise.

6.8.3.4 The heat path between the subsystem module housings and the support structure shall be designed to minimize heat transfer by conduction and to minimize the payload module thermal unbalance due to variations in the environmental temperature level.

6.8.3.5 Efficient thermal conduction paths shall be provided between the individual components to maximize the payload module housing heat sink capability.

6.8.3.6 An efficient conduction path between the ACS reaction subsystem and the entry shell shall be provided to maximize conductive heat exchange.

6.8.3.7 The interface between the Flight Capsule and the Flight Spacecraft shall be designed to minimize the transfer of thermal energy while in the Planetary Vehicle configuration.

6.8.3.8 The separation of the Flight Capsule from the Flight Spacecraft shall not perturb the heat balance of the Flight Spacecraft during its subsequent operations.

6.8.3.9 The Flight Spacecraft power subsystem shall supply the Flight Capsule electrical power for thermal control of the Flight Capsule while in the Planetary Vehicle configuration according to the following schedule.

Mission Phase	Approximate Time	Power Requirement
Near Earth cruise	190 days	100 watts
Near Mars cruise	190 days	150 watts
Mars orbit	3 to 10 days	210 watts
Preseparation	12 hours	230 watts

6.9 DATA ACQUISITION - ENGINEERING

The data acquisition - engineering primary functional area is separated into 13 distinct experiments to acquire the data required to fulfill the Capsule System (CS) engineering mission objectives. The specific functions, performance/design requirements, functional interfaces and component lists for each of these experiments are individually specified below.

6.9.1 Radiation Detector Experiment

6.9.1.1 Functions

The radiation detector's primary function is to detect and define trapped radiation belts in the vicinity of Mars. The radiation detector shall acquire data from separation to impact. The detector shall be designed primarily to detect low energy electrons but shall also be capable of detecting higher mass particles for comparison purposes.

6.9.1.2 Performance/Design Requirements

The radiation detector shall consist of four detectors. Two shall be capable of detecting electrons of greater than 40 kev. and protons of greater than 500 kev. The third detector shall be capable of detecting electrons of greater than 150 kev. and protons of greater than 3 Mev. The fourth detector shall be insensitive to electrons but shall detect protons between 500 kev. and 8 Mev. The count rate range of the detectors shall be between 0.2 and 3200 counts per second. The counts shall be accumulated for 5 seconds, read and dumped, and a new count interval started.

6.9.1.3 Functional Interfaces.

1. Electrical -- The radiation detection experiment shall require 0.8 watts at 28 volts DC

2. Mechanical -- The experiment weight shall not exceed 2.2 pounds nor the volume exceed 100 cubic inches. Mounting shall be with end windows having access to the outside of the vehicle with a look angle of 60 degrees with no structure obscuring a conical field of view. The end windows need not survive entry.

3. Thermal -- The experiment shall dissipate 0.8 watts internally during operation.

Operating temperature limits shall be +15 to 120°F.
Nonoperating temperature limits shall be -20 to 300°F.

4. Telecommunications -- Each of the four detectors shall provide one count output which shall be read and dumped once every 5 seconds. The accumulated count during that period shall vary from zero to 16,384 counts from each detector.

6.9.1.4 Component List

Components	Number Required
Low-energy electron detector (> 40 kev)	2
High-energy electron detector (> 150 kev)	1
Proton detector	1

6.9.2 Accelerometer Experiment

6.9.2.1 Functions

The accelerometers shall provide information related to the deceleration of the Entry Vehicle as it enters the Martian atmosphere. This data shall be used to correlate data on the entry trajectory, subsonic pressure, subsonic temperature, and altitude to reconstruct the atmospheric density profile. This experiment shall also provide data on the Entry Vehicle aerodynamic performance both during the entry and parachute descent phases of the mission. The accelerometers shall also provide some indication of the presence of high wind gusts as a backup to other wind velocity experiments.

6.9.2.2 Performance/Design Requirements

The experiment shall consist of three uniaxial accelerometers mounted with their respective axes normal to each other and parallel to the roll, pitch, and yaw axes. The array shall be located as close as possible to the Entry Vehicle center of gravity after the rocket motor has been expended. The accelerometers shall have a range of zero to 20 earth g in order to retain peak loading both during entry (4 - 17 g) and parachute opening (12 g) on scale. The accuracy of the measurement shall be ±0.1 percent of full scale. The measurement shall be made once each second from separation entry to impact.

6.9.2.3 Functional Interfaces

1. Electric Power and Control -- The accelerometers shall require 10.5 watts at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 1.8 pounds nor a volume of 15 in.^{3} (a cylinder of 1.75 inch length and 1.75 inch diameter), and shall be mounted as close as possible to the Entry Vehicle center of gravity with alignment of the units along the pitch, roll and yaw axes.

3. Thermal Control -- The experiment shall dissipate up to 10.5 watts internally during operation. Operating temperature limits shall be 15 to 120°F. Nonoperating temperature limits shall be -60 to 275°F.

4. Telecommunications -- Each of the three accelerometers shall provide one analog output which will be read with 10 bit accuracy once each second. Diagnostic and calibration data shall be limited to a null reading in a zero g environment.

6.9.2.4 Component List

Component	Number Required
Accelerometer	3

6.9.3 Mass Spectrometer Experiment

6.9.3.1 Functions

The mass spectrometer experiment shall provide information on the composition of the Martian atmosphere during the parachute descent phase of the mission.

6.9.3.2 Performance/Design Requirements

The experiment shall consist of a single quadrupole mass spectrometer operated in the scanning mode over the mass-to-charge range from 10 to 90 and with a resolution at m/e of 25 or better than a 1 percent adjacent peak contribution; that is, peaks of equal intensity at mass 25 and mass 26 should not cross contaminate each others central peak height by more than 1 percent. A multiplier type detector shall be utilized having a dynamic range of 10^4 with the least sensitive end of the range set to provide a full scale output with 10 millibars of carbon dioxide on the high pressure side of the leak. The scan time shall be less than one second. The detector's output logic shall provide two identifying signals for each such peak. One signal shall be the amplitude of the peak, logarithmically encoded, and the other, the time of the peak maximum from initiation of the scan. Each signal shall be transmitted in a 7 bit word. The system shall be capable of transmitting up to 80 peaks per scan every 5 seconds. The spectrometer shall receive samples of uncontaminated ambient Martian atmosphere flowing past the leak from parachute deployment to impact, and shall make measurements during this period.

6.9.3.3 Function Interfaces

1. Electrical Power and Control -- The mass spectrometer shall require 10 watts at 28 vdc.

2. Mechanical -- The weight of the experiment shall not exceed 10 pounds. The volume shall not exceed 265 in.^{3} contained in a single chassis of dimensions approximately 2 x 9.5 x 14 inches with its longest dimension parallel to the roll axis and with the forward end connected to the sampling manifold.

Alternatively, the experiment may be divided into four units for packaging convience. The units would have the following properties.

a. Analyzer Chassis -- weight 5 pounds, volume 112 in^3, 2 x 4 x 14 inches, maximum dimension parallel to roll axis, forward end connected to sampling manifold.

b. Ion Pump Chassis -- weight 1 pound volume 8 in^3, 2 x 2 x 2 inches, connected to rear of analyzer chassis.

c. Source Control Chassis -- weight 2.5 pounds, volume 72 in^3., dimensions not critical.

d. Detector Chassis -- weight 1.5 pounds, volume 72 in^3, dimensions not critical.

3. Thermal Control -- The mass spectrometer shall dissipate 10 watts internally during operation. Operating temperature limits shall be +15 to 120°F. Nonoperating temperature limits shall be -60 to 275°F.

4. Telecommunications -- As a result of a 1-second scan performed once every 5 seconds, the mass spectrometer shall provide up to 80 pairs of analog signals digitized as pairs of 7-bit words and transmitted once every 5 seconds. Additionally, measurements of total pressure, source temperature, and ionizing current shall be transmitted as diagnostic signals.

6.9.3.4 Component List

Component	Number Required
Mass spectrometer	1

6.9.4 Acoustic Densitometer Experiment

6.9.4.1 Functions

The acoustic densitometer experiment shall provide values of the velocity of sound, the acoustic impedance, and the temperature of samples of the Martian atmosphere for the purpose of determining density, mean molecular weight, and heat capacity ratio (C_p/C_v) of the atmosphere. Successive measurements shall allow construction of density and, to some extent, composition profiles with altitude. The experiment shall also provide functional redundancy with several of the other atmospheric measurements.

6.9.4.2 Performance/Design Requirements

The experiment shall consist of an acoustical chamber in the form of a tubular spiral with an acoustical generator at one end, and two microphones and a temperature measuring device along its length. The phase relationship between the signals at the two microphones shall be converted by onboard circuitry to represent the velocity of sound, while the amplitude of the signal at one microphone will be converted to represent the acoustic impedance. The data shall be encoded in 7 bit words for transmission. Preliminary phasing of the ranges shall be utilized to provide accuracies of the order of 1 percent.

The densitometer shall receive a continuous sample of uncontaminated ambient Martian atmosphere flowing past its aperture from parachute deployment to impact, and shall make measurements during this period.

6.9.4.3 Functional Interfaces

1. Electrical Power and Control -- The acoustic densitometer will require 4 watts at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 3 pounds nor a volume of 49 in^3, with dimensions of 7 x 7 x 1 inches, and shall be mounted adjacent to the atmospheric sampling manifold.

3. Thermal Control -- The densitometer shall dissipate 4 watts internally. Operating temperature limits shall be +15 to 120° F. Non-operating temperature limits shall be -60 to +275° F.

4. Telecommunications -- The experiment shall provide three analog outputs, read simultaneously with 7 bit accuracy once every 5 seconds.

6.9.4.4 Component List

Component	Number Required
Acoustic densitometer	1

6.9.5 Gas Chromatograph Experiment

6.9.5.1 Functions

The gas chromatograph experiment shall provide information on the amount of specific components of the Martian atmosphere during the parachute descent portion of the mission. This experiment shall provide functional redundancy with the mass spectrometer experiment.

6.9.5.2 Performance and Design Requirements

The experiment shall consist of an automatically programmed gas chromatograph to determine the amounts of argon, carbon dioxide, carbon monoxide, krypton, neon, nitrogen, oxygen and xenon present. The instrument shall collect a sample and carry out a complete analytical cycle every 5 seconds. The peak amplitudes for the various components shall be stored in preassigned channels and read every 5 seconds. The dynamic range and sensitivity will vary from channel to channel, as is shown below, but each shall be read with a 7 bit word.

Component	Range (partial pressure in mb)
Argon	0.05 to 5.0
Carbon Dioxide	1.0 to 15.0
Carbon Monoxide	0.0005 to 0.5
Krypton	0.005 to 0.5
Neon	0.05 to 1.0
Nitrogen	0.05 to 5.0
Oxygen	0.005 to 0.5
Xenon	0.005 to 0.5

6.9.5.3 Functional Interfaces

1. Electrical Power and Control -- The gas chromatograph will require 4 watts at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 5 pounds nor occupy a volume of more than 200 in^3, with dimensions of 5 x 5 x 10 inches. It shall be mounted immediately adjacent to the atmospheric sampling manifold.

3. Thermal Control -- The gas chromatograph shall dissipate 4 watts of electrical power internally during operation. It will probably generate an undetermined amount of heat chemically in a pulse prior to operation. Since the chemically generated heat will be used to maintain the chromatographic columns in an approximately isothermal state at a moderately elevated temperature (about 260° F) during the course of the experiment, the heated region shall be well insulated from the rest of the experiment and the rest of the payload. Thus this chemical heat shall not cause a significant thermal control problem prior to impact.

Operating temperature limits (outer surfaces) shall be +15 to +120° F. Nonoperating temperature limits shall be -60 to +275° F.

4. Telecommunications -- The experiment shall provide eight analog outputs read with 7 bit accuracy once every 5 seconds. In addition, there shall be two 7-bit diagnostic channels for column temperature and carrier gas pressure.

6.9.5.4 Component List

Component	Number Required
Gas chromotograph	1

6.9.6 Pressure Gage Experiment

6.9.6.1 Functions

The pressure gage experiment shall provide information on the atmospheric pressure during the lower velocity portions of the mission. The data so obtained shall not only be directly interpretable (with appropriate aerodynamic corrections) as atmospheric pressure, but will also allow estimation of the Entry Vehicle velocity. These lower altitude velocity data will be used in conjunction with the accelerometer and temperature data to provide an atmospheric density profile.

6.9.6.2 Performance/Design Requirements

The experiment shall consist of two pressure gages, one located at or near the front and the other at the rear of the suspended Capsule. These shall provide pressure data from which free stream velocities can be deduced with the help of analytical and test data. The range of pressures to be measured shall be from 0.02 to 20 millibars with an accuracy of 1 percent at the upper end of the scale and logarithmic encoding to provide the required dynamic range.

6.9.6.3 Functional Interfaces

1. Electrical Power and Control -- The pressure gage experiment shall require 3 watts at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 1.0 pounds nor a volume of 24 in^3, equally divided between two 2 x 2 x 3 inch gages. One of the gages shall be located near the front of the suspended capsule. The other gage shall be located on the back.

3. Thermal Control -- The experiment shall dissipate 3 watts internally. Operating temperature shall be +15 to +120°F. Nonoperating temperature shall be -60 to 275°F.

4. Telecommunications -- The pressure gage experiment shall provide two analog outputs to be read simultaneously once each second as two 7-bit works.

6.9.6.4 Component List

Component	Number Required
Pressure gage	2

6.9.7 Temperature Gage Experiment

6.9.7.1 Functions

The temperature gage experiment shall provide information on the temperature of the Martian atmosphere to be used in the reconstruction of the atmospheric density profile. It shall also provide information which will be useful in the design of the thermal control systems of future vehicles.

6.9.7.2 Performance/Design Requirements

The experiment shall consist of two mutually redundant temperature probes which provide total temperature measurements of the Martian atmosphere. With data provided from several other experiments (pressure, altimeters, acoustic densitometer, etc.) the static temperature will be computed. At lower velocities the total temperature will approach the static temperature. The probes shall have a temperature range of -235 to +275° F and an accuracy of ±2° F.

6.9.7.3 Functional Interfaces

1. Electric Power and Control -- The temperature gage experiment shall require 0.2 watt at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 0.6 pounds nor a volume of 8 in^3, equally divided between two probes about 1 x 1 x 4 inches. The probes shall be located behind the entry shell so as to make total temperature measurements from removal of the shell to impact.

3. Thermal Control -- The experiment shall dissipate 0.2 watt internally. Operating temperature of the sensing element shall be -235 to +275° F. Nonoperating temperature of the probe shall be -235 to +275° F.

4. Telecommunications -- The temperature gage experiment shall have two analog outputs read once each second formated as two 7-bit words.

6.9.7.4 Component List

Component	Number Required
Temperature probe	2

6.9.8 Beta Scattering Experiment

6.9.8.1 Functions

The beta scattering experiment shall provide information on the density of the Martian atmosphere from parachute deployment to impact. The experiment shall also provide functional redundancy on several other atmospheric experiments.

6.9.8.2 Performance/Design Requirements

The experiment shall consist of a radioioisotopic beta-ray source and beta detectors arranged so that the detectors respond to electrons back-scattered by the Martian atmosphere. The source strength and detector sensitivity shall be such that the experiment operates in the density range from 1×10^{-6} to 1×10^{-4} gm/cm^3. The experiment shall operate only during the parachute descent phase of the mission because of the difficulty of providing an adequate field of view prior to removal of the entry shell. The experiment shall accumulate counts for 1 second and then be read and dumped. A capability for accumulating up to 16,000 counts shall be included.

6.9.8.3 Functional Interfaces

1. Electric Power and Control -- The beta-scattering experiment shall require 0.3 watts at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 0.8 pounds nor a volume of 15 in.^{3} with dimensions of a cylinder of 3-inch diameter and 2-inch height. Mounting shall be such that one end of the cylinder shall have an unobstructed forward field of view, i.e., with no obstructions forward of the plane containing the end of the cylinder.

3. Thermal Control -- The experiment shall dissipate 0.3 watts internally during operation and will continuously dissipate an undetermined amount of power from radioactive decomposition from time of assembly. It is anticipated that the latter source of heat will not exceed 5 watts and probably will be considerably less than this.

6.9.8.4 Component List

Component	Number Required
Beta-scattering instrument	1

6.9.9 Water Detection Experiment

6.9.9.1 Functions

The water detection experiment shall provide information on the water vapor content of the Martian atmosphere during the parachute descent phase of the mission. The information obtained will be primarily of scientific interest and will assist in the planning of future biological and meteorological experiments.

6.9.9.2 Performance/Design Requirements

The water detector shall be capable of measuring partial pressures of water varying from 10^{-6} to 10^{-1} millibars over a temperature range of 150 to 300°K. The detector shall have an accuracy of ±30 percent of the amount present over the operating range. It shall provide logarithmic encoding of the data so that the results may be transmitted in a 7-bit word. The detector shall have a response time (90 percent of true reading) of less than 2 seconds over the entire operating range.

6.9.9.3 Functional Interfaces

1. Electric Power and Control -- The water detection experiment shall require 0.5 watt at 28 vdc.

2. Mechanical -- The experiment shall not exceed a weight of 0.5 pound, nor a volume of 16 in^3, and will be in the shape of a cylinder with a diameter of 1.5 inches and a length of 9 inches. It shall be mounted on the front of the suspended Capsule with its axis parallel to the roll axis and protruding ahead of the body of the suspended Capsule after jettisoning of the entry shell so that an unperturbed atmospheric sample may be obtained.

3. Thermal Control -- The experiment shall dissipate 0.5 watt internally during operation. Operating temperature limits shall be -189 to 80°F. Nonoperating temperature limits shall be -189 to +275°F.

4. Telecommunications -- The experiment shall provide one analog output read as a 7-bit word once each second.

6.9.9.4 Component List

Component	Number Required
Water detection experiment	1

6.9.10 Velocity-Attitude Sensor Experiment

6.9.10.1 Functions

The velocity-attitude sensor shall consist of a three-beam doppler radar. Its primary function shall be to estimate the horizontal velocity of the suspended Capsule during the parachute descent. It also shall have the capability of determining the suspended Capsule attitude. When used as an attitude sensor, the velocity-attitude sensor, provides a signal to initiate TV picture taking whenever the Suspended Capsule spin axis is within a preselected maximum cone angle with the local vertical. It shall also be capable of providing surface roughness information if the terrain is mountainous.

6.9.10.2 Performance/Design Requirements

The velocity-attitude sensor shall be capable of determining the horizontal velocity of the Suspended Capsule and its attitude to the local vertical with an accuracy of ±5 percent under conditions of Suspended Capsule tilt angles up to 45 degrees at altitudes ranging from 30,000 feet to impact.

6.9.10.3 Functional Interfaces

1. Electric Power and Control -- The velocity attitude sensor shall require 50 watts at 28 vdc.

2. Mechanical -- The experiment shall weigh 21 pounds. The two antennas shall require 1410 in^3 each, with a cylindrical base 15 inches in diameter and a height of 8 inches. The electronics requires a volume of 400 in^3.

3. Thermal Control -- The experiment shall dissipate 11 watts. Operating temperature limits shall be +15 to +120°F. Nonoperating temperature limits shall be -60 to +275°F.

4. Telecommunications -- There shall be 10 outputs of the velocity attitude sensor listed below.

Measurement	Number of Outputs
Range (3 beams)	3
Range rate (3 beams)	3
Doppler signal strength (3 beams)	3
Cone angle	1

These signals shall be processed via a 7-bit A-D converter.

6.9.10.4 Component List

Component	Number Required
Transceiver	1
Antenna	2

6.9.11 Television Experiment

6.9.11.1 Function

The television experiment shall provide high resolution images of the Martian surface during the parachute descent phase of the Flight Capsule mission.

6.9.11.2 Performance/Design Requirements

1. The television subsystem shall take television pictures of the Martian surface during parachute descent from 27,500 feet to impact. The television subsystem shall produce as many images as possible (but at least nine images) consistent with the following major constraints:

   a. All picture data shall be transmitted prior to impact.

   b. A nominal 15,000 k bits/sec transmission rate shall be available for TV data.

2. The television image format shall consist of 200 lines x 200 electrically resolvable picture elements. Each image element shall be quantized to at least 32 grey levels.

3. Three types of images are required:

   a. Wide angle - low resolution images.

   b. Medium angle - medium resolution images.

   c. Narrow angle - high resolution images.

4. Fixed focus and focal length cameras shall be used. Three cameras shall satisfy the image requirements above.

5. The nominal resolution and field of view specifications for the three cameras are as follows:

Camera	Field of View (degrees)	Resolution (for vertical image at 20,000 feet altitude)
A-Camera	24.0	30 ft/TV line
B-Camera	8.2	10 ft/TV line
C-Camera	2.7	3.3 ft/TV line

6. At least one image is required at lower altitude to yield 1 ft/TV line resolution.

7. Nested images covering a 9:3:1 resolution range are required in connection with coarse resolution from 100 to 10 ft/TV line. The required nesting shall be achieved by simultaneously exposing the three cameras with their optical axes oriented to contain the C-camera image within the B-camera image and the B-camera image within the A-camera image.

8. The cameras shall operate over a range of surface background illumination from 30 ft-Lamberts with possible highlights to 3000 ft-Lamberts and be designed to achieve unity signal-to-noise ratio (SNR) at brightness levels of 30 ft-Lamberts. Unity signal-to-noise ratio is defined in this context as the condition when vidicon signal is equal to the peak-to-peak noise.

9. The television images shall cover the visible spectrum from 4000 A to 7000 A with a peak spectral response between 4800 A and 5800 A.

10. Two color filters shall be employed on the low resolution A-camera. Filter spectral response shall be chosen such that the combined spectral response of the vidicon and filters results in equal energy transmission on each portion of the filtered image. Peak response of the filter-vidicon combinations shall be between 4700 A to 5500 A and 5800 A to 6200 A, respectively.

11. Shutter durations shall be selected to provide sufficient light energy to the vidicons and, with the camera stabilization system, assure that smear due to Flight Capsule dynamics is less than 1/2 of a television line for each camera at all anticipated deployment altitudes. All camera shutters shall be activated such that the shorter shutter openings occur during the longer shutter openings.

12. The camera stabilization subsystem shall compensate for Flight Capsule motions in pitch and yaw during the picture exposure intervals, but is not required to compensate image motion due to drift velocity over the surface, descent velocity normal to the surface, or Flight Capsule roll.

13. The TV subsystem shall be packaged and/or housed in a container which:

   a. Protects the optical elements and image tubes from thermal gradients encountered during the Flight Capsule mission.

b. Maintains operating temperatures between +20 and +100° F and storage temperatures between 0 and +140° F.

c. Prevents arcing resulting from high voltage requirements of the television cameras.

6.9.11.3 Functional Interfaces

1. Electrical -- The television subsystem shall require 27 watts at 28 vdc while operating and 5 watts intermittently at 28 vdc while in a nonoperating state.

The CC&S system shall control the television experiment through the following commands:

a. Checkout (prior to or during flight)

b. Calibrate (prior to or during flight)

c. Release Gimbals (after the parachute has been deployed)

d. Open Shutter (at appropriate times during the parachute descent phase).

The release gimbals and open shutter commands shall be supplied in accordance with the logic indicated in Figure 12.

In addition, the CC&S system shall provide a standard clock pulse at any convenient frequency between 1 and 100 kilocycles for internal timing of the TV system.

The TV system shall provide two signals to the CC&S system:

a. Shutter Confirmation (when the shutters are fired).

b. Gimbal Release Confirmation (indicating that the gimbals are both mechanically free to respond to IRS commands).

The attitude control subsystem shall provide Flight Capsule attitude data and develop control signals required for camera stabilization as follows:

a. The IRS computer shall provide digital indications of the Suspended Capsule roll axis look angle with respect to the vertical, $\dot{\theta}_c$, and the Suspended Capsule swing rate, θ_c , as follows:

$|\theta_c| \leq 30 + 3°$

$|\dot{\theta}_c| \leq 13$ degree/sec + 1 degree/sec

b. The IRS shall provide gimbal control signals required for camera stabilization. A two axis gimbal system is required to stabilize the camera in pitch and yaw. Roll stabilization is not required. The gimbal control signals shall result in maintaining the camera optical axis within ±1 degree of local vertical and reducing camera pitch and yaw rates to less than 0.001 deg/sec. The gimbal control signals shall result in this performance for Suspended Capsule roll axis within ±45 degrees of local vertical.

The diagnostic data acquisition subsystem shall provide sampling analog and event data required to assess the operation of the television system.

2. Mechanical -- The television subsystem shall not exceed a weight of 60 pounds nor a volume of 2080 in^3.

The separation subsystem shall provide a viewing port of sufficient size to avoid obstructing the field of view of the television cameras at altitudes lower than 25,000 feet and Mach numbers lower than 1.2.

In the primary operating mode (with heat shield and entry shell separated) the camera viewing port shall include at least 52 degrees of a solid angle centered around the vehicle roll axis and pointing downward. In the backup mode (heat shield and entry shell attached) a viewing port solid angle of at least 45 degrees is required.

3. Thermal -- The television subsystem shall dissipate 27 watts internally during operation. Operating temperature limits shall be +20 to +100°F except for portion of the TV subsystem exposed to the Martian atmosphere during parachute descent. Nonoperating temperature limits shall be 0 to +140°F excluding sterilization.

4. Telecommunication -- The telecommunications system shall provide storage and transmission of digital video and video identification data. The major functional interface between the TV experiment and telecommunications is shown in Figure 13.

Analog-to-digital converters in the TV subsystem shall supply parallel 5-bit video signals to digital memories in the telecommunications system. Each camera shall generate 40,000 5-bit words in 6 second readout periods which occur at least 28.4 seconds apart. These data shall be generated in synchronism with a read synchronization signal supplied by telecommunications. The telecommunications system must insure that pictures are transmitted in the order CBA CBA CBA----------.

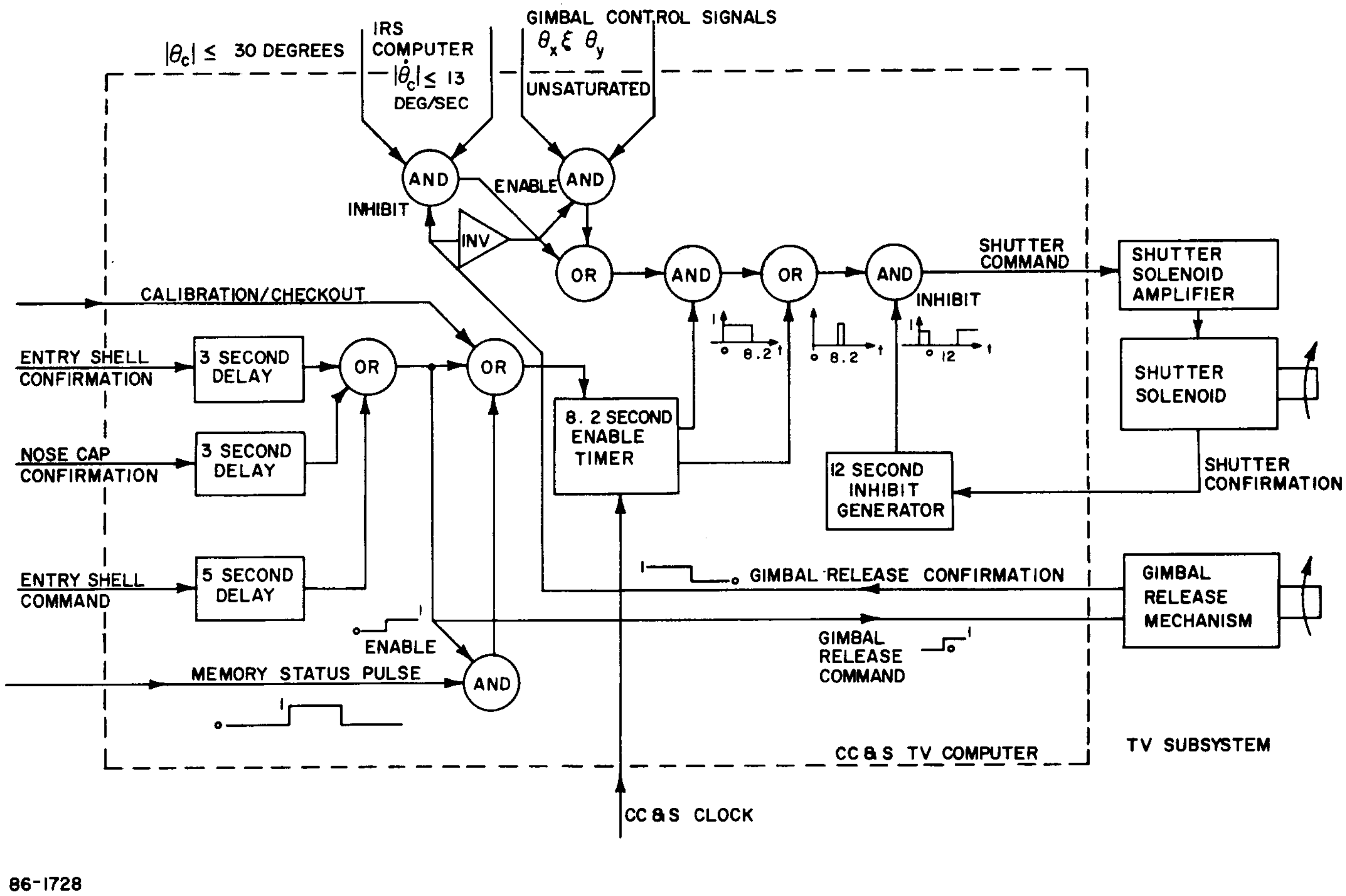

Figure 12 TV COMPUTER AND SHUTTER LOGIC

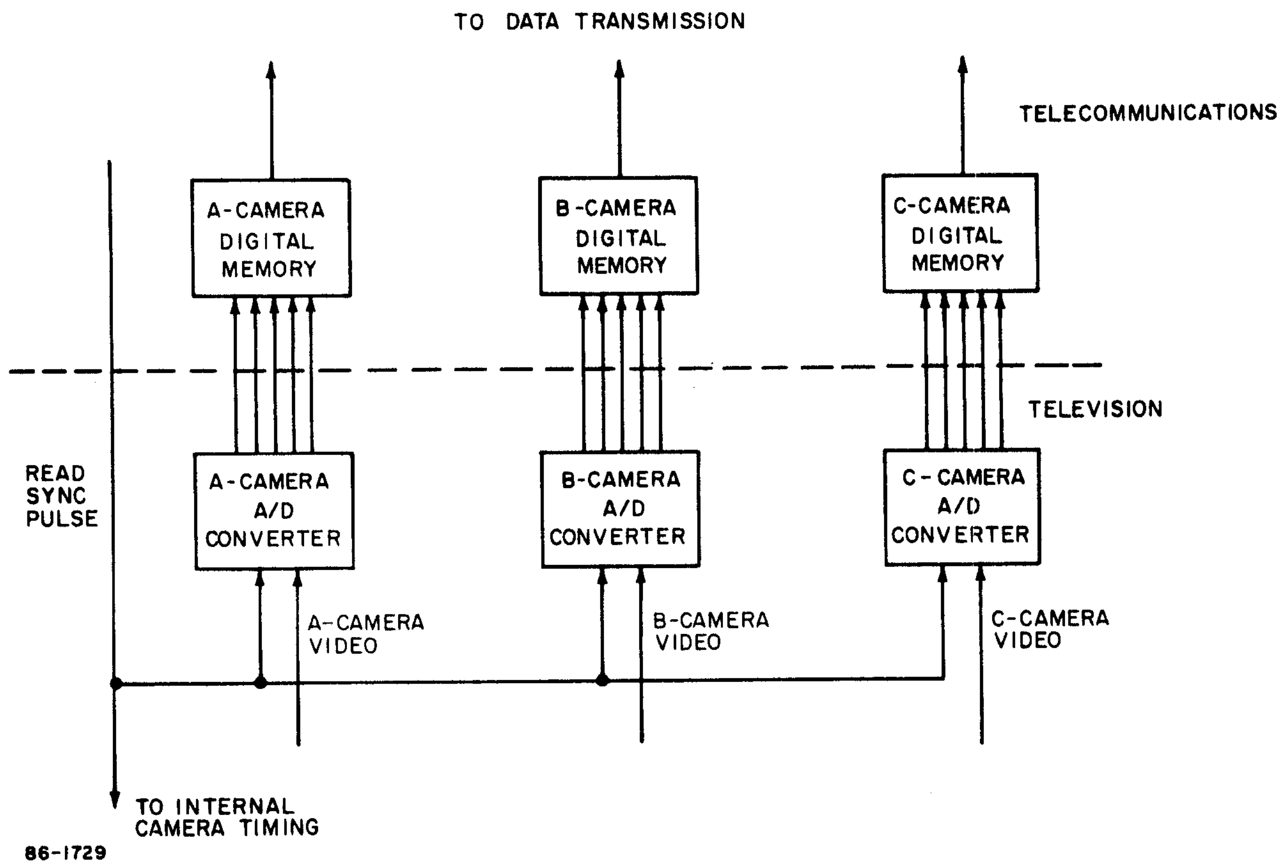

Figure 13 TV - TELECOMMUNICATIONS INTERFACE

6.9.11.4 Components List

Component	Quantity
Television subsystem	1

6.9.12 Penetrometer Experiment

6.9.12.1 Functions

The penetrometer experiment shall provide information on the physical nature of the Martian surface to allow estimates to be made of the bearing strength of the surface and of the vertical structure of the near-surface region. The experiment shall be designed to provide this data at four different sampling points. The exact distance between the sampling points will depend on the wind velocities and descent dynamics encountered.

6.9.12.2 Performance/Design Requirements

The experiment shall consist of four penetrometers to be dropped sequentially from the Suspended Capsule. The first drop shall be controlled by the radar altimeter at 3500 ± 175 feet, the remaining penetrometers following at 2-second intervals. Each penetrometer payload shall consist of an omnidirectional accelerometer and a transmitter with an internal power supply. Each penetrometer shall transmit a single analog output to the receiver on the Suspended Capsule which shall be capable of receiving all four transmissions simultaneously. The analog inputs shall be converted to up to 50 discrete data points on the deceleration-time curve. The four penetrometer transmitters shall operate on four channels spaced at 264 kilocycles intervals around 430 megacycles. The penetrometer payloads shall be surrounded by a shell of balsa impact limiter to allow a maximum impact velocity of 325 ft/sec with maximum impact loading of 10,000 g.

6.9.12.3 Functional Interfaces

1. <u>Electrical Power and Control</u> -- The penetrometers shall carry their own internal power supplies. The receiver shall require no more than __*__ watts at 28 vdc.

2. <u>Mechanical</u> -- Each penetrometer shall not exceed a weight of 9 pounds nor require a spherical volume of more than 1240 in.^{3} with approximately a 13.3 inch diameter. The receiver shall not exceed a weight of 5 pounds nor require a volume of more than 150 in^3. The penetrometers shall be mounted in the forward part of the Suspended Capsule in such a manner that they fall free from the Suspended Capsule when released.

3. Thermal Control -- The penetrometers shall be in a nonoperating mode while aboard the Flight Capsule and therefore, dissipate no heat. The receiver shall dissipate 5 watts internally.

Operating temperature limits for the receiver shall be +15 to +120°F. Nonoperating temperature limits shall be -60 to +275°F.

4. Telecommunications -- The penetrometer receiver shall provide the telecommunications subsystem with separate analog outputs for each penetrometer. The telecommunications shall be capable of digitizing the data to provide 15 preselected g-levels and the time between levels to the nearest 0.1 msec. The time between levels may be as much as 6 msec (6 bits).

6.9.12.4 Component List

Component	Number Required
Penetrometer	4
Receiver	1

6.9.13 Radar Altimeter Experiment

6.9.13.1 Functions

The radar altimeter experiment shall consist of a high altitude and a low altitude altimeter. The high-altitude altimeter shall operate until entry heat shield jettison and the low-altitude altimeter from heat shield jettison to near impact. During the high altitude phase, the altimeter shall provide altitude data for various scientific measurements, and parachute deployment. During the low altitude phase, the radar shall provide altitude data for dropping penetrometers.

A secondary objective of both phases shall be to permit the estimation of terrain roughness by spectral analysis of the returned signals. Also during the low altitude phase, an estimation of the Suspended Capsule horizontal velocity will be made from spectral analysis of the return. The low-altitude antenna shall also be used as the penetrometer antenna.

6.9.13.2 Performance/Design Requirements

The high altitude altimeter shall be capable of measuring altitude from 250,000 to 18,000 feet with an accuracy of ± 5 percent when the Suspended Capsule axis is within 70 degrees of the local vertical. The low-altitude altimeter shall be capable of measuring altitude from 30,000 to 100 feet with an accuracy of 5 percent when the Flight Spacecraft axis is within 70 degrees of the vertical.

6.9.13.3 Functional Interfaces

1. Electric Power and Control -- The high-altitude radar shall require 60 watts at 28 vdc. The low-altitude radar shall require 4 watts at 28 vdc. The altimeter shall be fused and incorporate diodes to accept power from two sources (see Section 6.10).

2. Mechanical -- The electronics module weight shall not exceed 11 pounds and volume shall not exceed 300 in^3. The volume of the low-altitude antenna shall not exceed 4180 in^3. The weight of the low-altitude antenna shall not exceed 7 pounds and the weight of the diplexer shall not exceed 1.5 pounds. The high-altitude antenna shall be part of the Suspended Capsule structure.

3. Thermal Control -- The high- and low-altitude altimeters together shall dissipate about 35 watts. Operating temperature limits shall be +15 to 120°F. Nonoperating temperature limits shall be -60 to +275°F.

4. Telecommunications -- Continuous altitude information shall be supplied to the telecommunications subsystem by means of 7-bit analog-to-digital converter for correlation with the science measurements. Also during discrete intervals of both the high and low altitude phase, the return signal time waveform shall be transmitted to the A-D converter.

6.9.13.4 Component List

Component	Number Required
Transceiver	1
High-altitude antenna	1
Low-altitude antenna	1
Diplexer	1

6.10 ELECTRICAL POWER AND CONTROL

6.10.1 Functions

The primary function of the electrical power and control subsystem shall be to provide an electrical energy source whose output is appropriately conditioned and controlled to the requirements of the Flight Capsule subsystems. As a secondary function, it shall be capable of receiving and conditioning electrical energy from an external Flight Spacecraft or OSE source, to supplement its primary function and to maintain a specified level of stored electrical energy during all mission phases prior to separation from the Flight Spacecraft.

6.10.2 Performance/Design Requirements

6.10.2.1 Design Description

The power and control subsystem shall consist of two batteries, two voltage regulators, two power controls and two battery chargers. The parts of the subsystem shall be interconnected in such manner to minimize the effects of failure of any component in the power subsystem.

The power control unit shall include the switches, diodes, and fuses shown on Figure 14. These fuses and diodes shall be connected such that only one power line is necessary from the power control units to each component except in the cases where the diodes and fuses are indicated to be within the component. In these cases, two power lines, one from each of the power busses, shall supply the component. Fuses shall be utilized for circuit protection in the event of a short circuit failure of any component. Each component shall be powered simultaneously from battery 1 and battery 2. Diodes shall be incorporated to prevent a short circuit in one distribution system from draining the other source.

The control section shall consist of silicon control rectifiers arranged to perform switching of power on receipt of proper signals from the Flight Spacecraft and the Flight Capsule programming and sequencing subsystems.

In addition to the main power and control subsystem, each penetrometer shall contain a rechargeable battery which shall be maintained in a charged condition by the battery chargers.

The battery chargers shall be located in the Flight Spacecraft.

6.10.2.2 Design Operation

The operation of the subsystem shall take place in three modes:

1. Cruise to Mars encounter,
2. Preseparation checkout, and
3. Separation to impact.

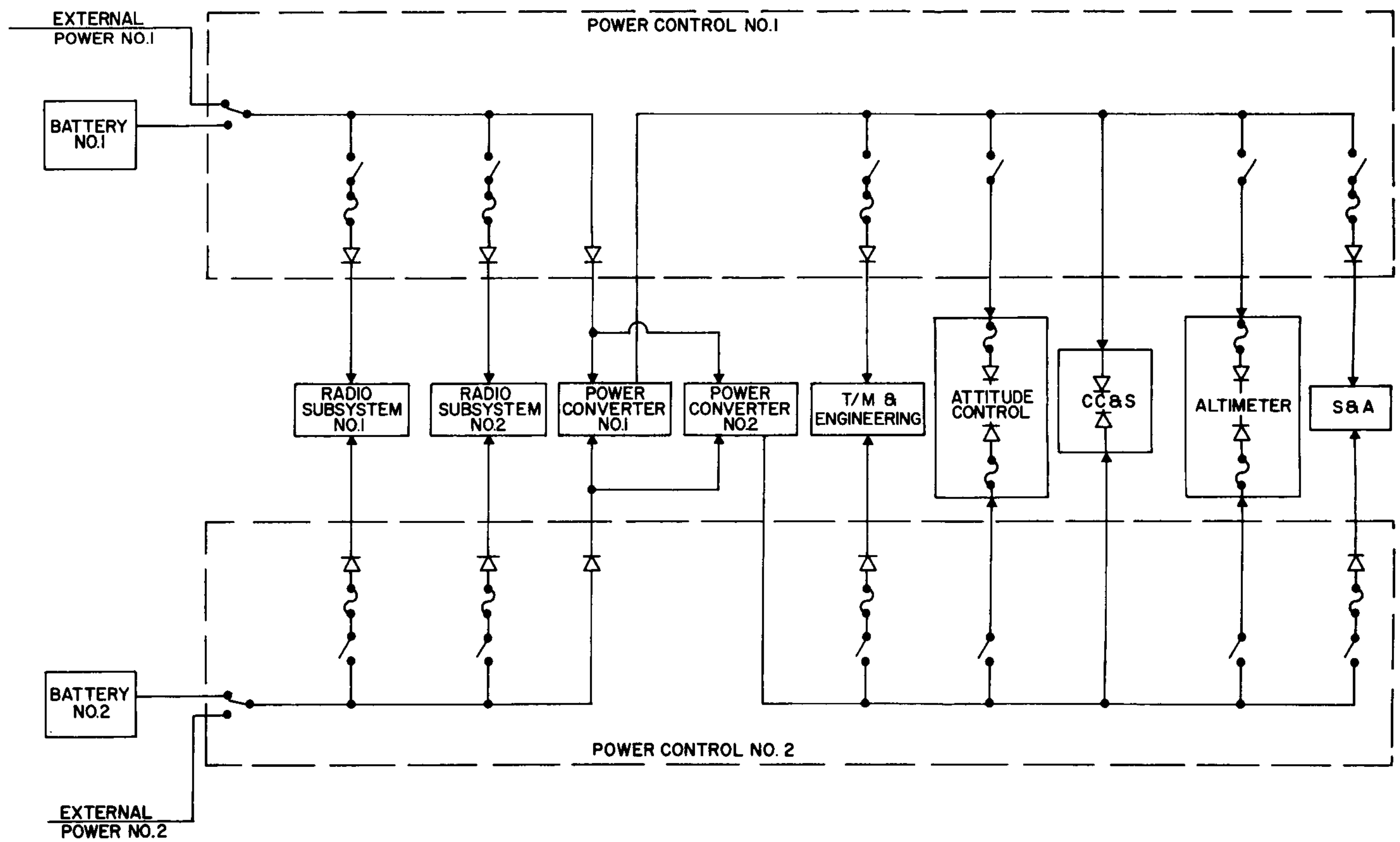

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Figure 14 POWER DISTRIBUTION SUBSYSTEM

1. Cruise to Encounter -- A portion of the Flight Spacecraft solar panel power output shall be continuously furnished to the Flight Capsule for active temperature control and battery trickle charge, in addition to periodic requirements for subsystem status checkouts.

2. Preseparation Checkout -- Power shall be supplied by the Flight Capsule batteries to the Flight Capsule communications and instrumentation in order to check Flight Capsule status. During the checkout period all power used by the Flight Capsule shall be supplied by the Flight Capsule batteries.

Preseparation checkout shall be completed at a sufficient time before separation to permit fully recharging the Flight Capsule batteries.

3. Separation through Impact -- After separation from the Flight Spacecraft the Flight Capsule battery-regulator subsystem shall supply power to all Flight Capsule subsystems.

6.10.2.3 Performance/Design Parameters

1. Battery

a. Performance-Single System -- Minimum rated capacity of 740 watt-hours at +40°F discharge temperature for each battery. Cells shall be hermetically sealed nickel cadmium.

Output Voltage: 20 to 32 vdc under load

28.0 volts, plateau

35.0 volts, open circuit

Maximum Load: 550 watts

b. Temperature Limits

Storage (nonoperating - charged): 100 to +200°F,

Storage (nonoperating - discharged): 100 to +300°F,

Operating (launch to separation): +40 to +200°F,

Average charge efficiency: 70 percent.

c. Battery Design Summary - Each battery

Weight: 53 pounds

Volume: 530 in^3

Dimensions: 10 inches minimum on one dimension, may be distributed as single cells connected by harness or packaged in boxes as required. Each cell shall be hermetically sealed in steel case.

2. Voltage Regulator - Each Regulator

a. Performance -- The voltage regulator shall provide power for the component power users listed in Table XVII.

Input: 20 to 35 vdc

Voltage Output: 28 vdc ± 1 percent

Ripple: Less than 0.10 volts peak-to-peak

Transients: less than 0.5 volts for 5 amp load changes

Current Output: 0.5 to 20 amp continuous, up to 25 amp intermittent.

b. Temperature

Nonoperating: - 40 to +300°F

Operating: +40 to +160°F (to 200°F during transients)

c. Voltage/Regulator Design Summary - Each

Minimum Conversion Efficiency: 75 percent

Semiconductors: all silicon

Dimensions: 6 1/4 x 8 x 4 inch

Weight: 7 pounds

3. Battery Charger (Location: -Flight Spacecraft)

a. Performance

Inputs: 33 to 52 vdc

Outputs - two - each with the following characteristics

Trickle at 0. 2 to 0. 3 amps

Charge rate: 2 amps nominal

Charging voltage: 28 to 36 vdc.

b. Charger Design

Weight: 3. 0 pounds

Volume: 120 in^3

4. <u>Power Control - Each Control</u>

a. Performance -- The power control shall provide the switch functions shown on Table XVIII, according to the following criteria:

Rated nominal power handling: 550 watts

Turn on time for low power users: 2 sec with 3 volt signal

Turn on time for high power users: 3 sec with 3 volt signal

Nominal firing signal (one switch): 2 volts, at 30 ma.

Total switches (S. C. R's): 13 low power (1. 1 amp max.)

24 high power (5. 5 amp max.)

Fuse rating - slow blow type: 200 percent of expected load in that circuit.

Maximum heat dissipation: 27 watts

Maximum forward voltage drops:

Low power - 1. 1 volts at 1. 1 amp.

High power - 0. 8 volts at 5. 5 amp.

b. Temperature Environment

Nonoperating: - 85 to 300°F

Operating: - 85 to +255°F

c. Power Control Design Each System

Size: 4 x 6 1/4 x 6 1/2 inches

Weight: 8.6 pounds

6.10.3 Functional Interfaces

6.10.3.1 Electrical

1. Inputs to the Electrical Power and Control Subsystem

a. From the Launch Complex Equipment (LCE)--The electrical power and control subsystem shall receive: External power to operate the Flight Capsule subsystem during prelaunch checkout at 28 vdc nominal.

External power to keep the batteries charged at 15 milliamps per amp-hour of battery capacity for the trickle charge and the current equivalent of the 16-hour rate of full charge.

b. From the Flight Spacecraft Photovoltaic-- Solar Panel Power Supply--The electrical power and control subsystem shall receive a dc input to charge the battery and supply environmental control.

c. From the CC & S -- The electrical power and control subsystem shall receive signals to fire the SCR switches.

2. Outputs from the Electrical Power and Control Subsystem

a. To the Launch Complex Equipment -- Indications and measurements for prelaunch monitoring and subsystem performance.

b. To Telemetering Subsystem -- A number of analog signals representing measurements of power subsystem operating parameters. Specific measurements shall be made for both prelaunch and post-launch as shown in paragraph 6.11.2.3. Each output will be conditioned to the standard telemetry measurement range of 0 to 5 volts.

TABLE XVII

COMPONENT POWER USERS

Component	Average Power (watts)
Transmitters (1)	170
Engineering data handling	6
Diagnostic data handling	4
Data storage	7
Delay data storage	2
Doppler radar (2)	10/50
Mass spectrometer	10
Radiation detector	0.8
Accelerometer	10.8
Acoustic densometer	4
Gas chromatrograph	4
Pressure sensor	21.1
Beta scatter	0.3
Temperature sensors	7.7
Radar altimeter (3)	60/4
Vibration	0.5
Ablation	0.8
Penetrometer receiver	5.0
Water detector	0.5
Television (2)	5/27
ACS electronics	10
Inertial reference subsystem	45
Sentry gyro	10
Voltage regulator	86
	480.5/486.5

1. All power except that delivered to the transmitters and voltage regulator shall be regulated at 28 vdc ± 1 percent by the power convertor/regulator. Power delivered to the transmitter and regulator shall be between 20 and 35 vdc.

2. Doppler radar standby power is 10 watts; full power 50 watts after entry shell deployment. TV standby power is 5 watts; full power is 27 watts.

3. High-altitude altimeter power is 60 watts; low-altitude altimeter is 4 watts.

TABLE XVIII

POWER CONTROL FUNCTIONS

Destination of Command	Function	Purpose
Sterile canister S&A device	Safe/arm	Permit firing pyrotechnics associated with the sterile canister lid ejection
Sterilization canister depressurizing valve	Vent	Reduce pressure prior to lid removal
Sterilization canister lid pyrotechnics	Fire	Remove lid
Altimeter	Low/high	Switch between high-altitude and low-altitude altimeter
Attitude control	On/off	
Telecommunications 1	On/off	
Telecommunications 2	On/off	
Engineering Experiments 1	On/off	
Engineering Experiments 2	On/off	
Separation event S&A device	Safe/arm	Permit firing pyrotechnics associated with separation
Open and gas samples	Release	Provide samples of gas to calibrate mass spectrometer gas chromatograph and densitometer
Fire separation squibs	Fire	Separate FC from FS
Battery 1	Off/charge	
Battery 2	Off/charge	
Cold gas supply valve 1 & 2	Open	Permit cold gas to flow to nozzles
Thrust vector control (TVC) igniter	Ignite	
Rocket igniter	Ignite	
Backup TVC igniter	Ignite	
Backup rocket igniter	Ignite	
Power convertors	Internal/External	To select power source
Pilot parachute deployment squibs	Fire	
Backup pilot parachute squibs	Fire	
Main parachute deployment squibs	Fire	
Entry shell attachment squibs	Fire	
First penetrometer ejection squibs	Fire	
Second penetrometer ejection squibs	Fire	
Third penetrometer ejection squibs	Fire	
Fourth penetrometer ejection squibs	Fire	

6.10.3.2 Mechanical

The batteries and discharge-regulators shall be attached to the Flight Capsule structure. The penetrometer batteries shall be attached to the penetrometer support structure. The batteries shall be connected by cable to the Flight Spacecraft where the charge regulators are located. This link shall be broken at separation.

6.10.4 Component List

Component	Quantity Required
Battery charger	2
Battery	2
Voltage regulator	2
Power control	2

6.11 TELECOMMUNICATIONS

6.11.1 Functions

The overall functional requirement for the telecommunication subsystem shall be to telemeter certain diagnostic and experimental data from the Flight Capsule to the Flight Spacecraft. The overall telecommunication subsystem shall be comprised of the following listed subsystems:

1. FC radio subsystem
2. FC data handling subsystem
3. FS relay radio subsystem
4. FS relay data handling subsystem

6.11.1.1 FC Radio Subsystem

The Flight Capsule radio subsystem shall perform the following listed functions:

1. Modulate the transmitted RF signal with a telemetry signal.
2. Transmit a modulated RF signal to the Flight Spacecraft.

6.11.1.2 FC Data Handling Subsystem

The Flight Capsule data handling subsystem (DHS) shall perform the following listed functions:

1. Provide data handling and storage services for the Flight Capsule diagnostic and experimental instrumentation.

2. Time multiplex (commutate) diagnostic and experimental signals.

3. Condition certain diagnostic and experiment signals so as to match their characteristics to the respective encoding circuits.

4. Convert each commutated diagnostic data sample to a 7-bit binary word.

5. Control and synchronize the experiment instruments so that the instrument internal sequence is known.

6. Provide the necessary sampling rates, both simultaneous and variously sequential, to ensure meaningful experiment data.

7. Perform the necessary conversions and encoding of the several forms of experiment data and place them in a suitable format.

8. Buffer the experimental data, which occur at different and sporadic rates, and make them available at various, but constant, desired rates to the time multiplexer.

9. Use the binary coded and time multiplexed diagnostic and experiment data to frequency-shift-key (FSK) modulate the RF carrier of the Flight Capsule to Flight Spacecraft telemetry link.

10. Provide a time delayed (stored) replica of certain diagnostic and experiment data (in binary form) for inclusion in the time multiplexed sequence.

6.11.1.3 FS Relay Radio Subsystem

The FS relay radio subsystem shall perform the following listed functions:

1. Noncoherently receive and demodulate a FSK RF signal from the Flight Capsule.

2. Detect the demodulated signal and reconstruct, within a specified error rate, the binary waveform originating in the Flight Capsule data handling subsystem.

3. Transfer the reconstructed binary waveform signal to the Flight Spacecraft relay data handling subsystem.

6.11.1.4 FS Relay Data Handling Subsystem

The Flight Spacecraft relay data handling subsystem shall perform the following listed functions:

1. Store the data received by the Flight Spacecraft relay radio subsystem.

2. Provide the stored data to the Flight Spacecraft data handling subsystem at the time, at the rate, in the form required.

6.11.2 Performance/Design Requirements

The values of the principle parameters that determine overall subsystem performance shall be as stated below:

6.11.2.1 FC Radio Subsystem Parameters

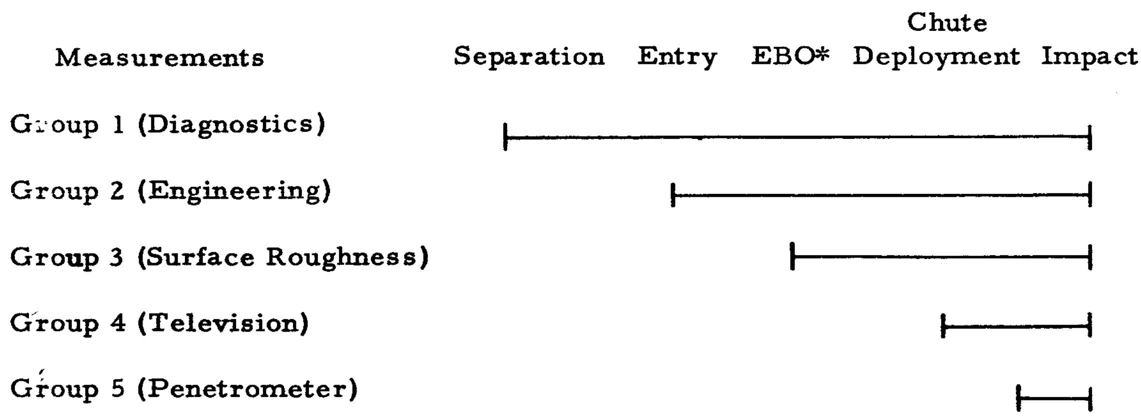
1.	Carrier frequency	267-273 mc band
2.	Radiated power	30 watts (minimum)
3.	Frequency stability	±0.005 percent
4.	Modulation	FSK
5.	Frequency deviation	± 100,000 cps
6.	Antenna gain	-14 db (minimum in 270 degree beamwidth)
7.	Antenna beamwidth	270 degrees (minimum)
8.	Antenna polarization	circular

6.11.2.2 FS Relay Radio Subsystem Parameters

1.	Carrier frequency	267-273 mc band
2.	Noise figure	4 db maximum
3.	Demodulation	Noncoherent FSK
4.	Threshold sensitivity	-172 dbm (maximum)
5.	Antenna gain	2 db (minimum) in 130 degrees beamwidth
6.	Antenna beamwidth	130 degrees (minimum)
7.	Antenna polarization	Circular

6.11.2.3 FC Data Handling Subsystem Parameters

1.	Data sampling requirements	See Figure 15 and Table XIX
2.	Data format output	Pulse-code-modulation (PCM), serial-binary-coded, time division multiplexed samples.
3.	Output data rate	18,000 bps (maximum)
4.	Data frame length	1064 bits
5.	Data frame synchronization	31-bit PN-code word
6.	Data word length	7-bit nominal
7.	Data storage	Data samples collected during blackout (Group 2 of Table XIX shall be stored for 100 seconds then retransmitted prior to impact.)



*End of Blackout

Figure 15 DATA COLLECTION PROFILE

TABLE XIX

DATA REQUIREMENTS SUMMARY

Measurements	Inputs		Sampling Rate	Accuracy
	Analog	Digital		
Group 1	109		0.2 sps	5 percent
	4		0.2 sps	2 percent
	68		0.2 sps	1 percent
	7		0.2 sps	0.1 percent
		12 events	0.2 sps	2 second
		35 events	0.2 sps	1 second
		21 events	0.2 sps	0.5 second
Group 2	171		0.2 sps	1 percent
		4 (count)	0.2 sps	7 bits
	9		1.0 sps	0.1 percent
	15		1.0 sps	1 percent
		3	1.0 sps	7 bits
		1 (count)	1.0 sps	14 bits
Group 3		1	150 sps	1 percent
Group 4		3	7.5 kc/s	5 bits
Group 5	4		10 kc/s	0.1 msec

8. Time Diversity — All data collected during parachute descent shall be transmitted twice with a time delay between transmissions of approximately 2.5 seconds.

9. Data List — Tables XX and XXI list all data expected for a typical mission.

6.11.2.1 Additional Performance Requirements and Constraints

1. Requirements -- The telecommunication subsystems shall meet the radio frequency interference requirements (RFI) and all other environmental and general design requirements included as part of this specification.

2. Performance Margin -- Telemetry link performance margin shall be plotted as a function of time in minutes from separation.

3. Performance Definitions -- The following definitions shall apply to the overall performance curves:

a. Performance Margin -- The performance margin is defined as the ratio of the nominal received signal level to the nominal threshold signal level expressed in db. It shall be considered acceptable when the margin is equal to or greater than the linear sum of the adverse system tolerances.

b. Nominal Received Signal Level -- The nominal received signal level is defined as the received signal level as calculated from the nominal system parameters (gains, losses, and power levels). The calculation excludes as far as possible all arbitrary margins, pads, or unknown factors.

c. Nominal Threshold Signal Level -- The nominal threshold signal level shall be the received signal level required to achieve a specified threshold signal-to-noise ratio in the effective noise bandwidth of the demodulator given the system noise spectral density.

d. Threshold Signal-To-Noise -- The threshold signal-to-noise power ratio shall be the signal-to-noise ratio required at the detector that will result in the minimum acceptable system performance.

TABLE XX

ENGINEERING DATA REQUIREMENT LIST

Measurement	Origin	Sample Rate	Accuracy
1. Pressure 1	PT 1	1 sps	1%
2. Pressure 2	PT 2	1 sps	1%
3. Radiation (4 outputs)	Radiation detector	0.2 sps	
4. X axis acceleration	Accelerometer ACS	1 sps	0.1%
5. Y axis acceleration	Accelerometer platform	1 sps	0.1%
6. Z axis acceleration	Accelerometer mounted	1 sps	0.1%
7. Temperature 1	TC 1	1 sps	1%
8. Temperature 2	TC 2	1 sps	
9. Argon content	Gas chromatograph	0.2 sps	
10. Carbon dioxide content		0.2 sps	
11. Nitrogen content		0.2 sps	
12. Carbon monoxide content		0.2 sps	
13. Oxygen content		0.2 sps	
14. Water content		0.2 sps	
15. Neon content		0.2 sps	
16. Krypton content	Gas chromatograph	0.2 sps	
17. Temperature	Acoustic densitometer	0.2 sps	1%
18. Velocity	Acoustic densitometer	0.2 sps	1%
19. Impedance	Acoustic densitometer	0.2 sps	
20. Beta scatter	Beta scatterer	1 sps	
21. Water content	Water detector	1 sps	1%
22. Surface roughness	Radar altimeter	150 sps	1%
23. Altitude		1 sps	
24. Altitude rate	Radar altimeter		
25. Cone angle	Doppler radar		
26. Doppler signal strength 1			
27. Doppler range 1			
28. Doppler range rate 1			
29. Doppler signal strength 2			
30. Doppler range 2			
31. Doppler range rate 2			
32. Doppler signal strength 3			
33. Doppler range 3			
34. Doppler range rate 3	Doppler radar	1 sps	
35.-114. Mass. channels 1 through 80	Mass specifications	0.2 sps	1%
115. Surface hardness 1	Penetrometer receiver	10 ksps from 3000' to impact	±0.1 ms
116. Surface hardness 2			
117. Surface hardness 3	Penetrometer reciever	10 ksps from 3000' to impact	±0.1 ms
118. Television 1	A-camera	7.5 ksps	3%
119. Television 2	B-camera		
120. Television 3	C-camera	7.5 ksps	3%
121. Claibration			0.1%
122. X axis acceleration	Accelerometer body	1 sps	0.1%
123. Y axis acceleration	Accelerometer mounted	1 sps	0.1%
124. Z axis acceleration	Accelerometer	1 sps	0.1%

TABLE XXI

DIAGNOSTIC DATA REQUIREMENTS LIST

Measurement	Sample Rate (SPS)	Accuracy
1. Mass spectrometer temperature	0.2	5%
2. Mass spectrometer pressure	0.2	5%
3. Mass spectrometer current	0.2	5%
4. Gas chromatograph temperature	0.2	5%
5. Gas chromatograph pressure	0.2	5%
6. Penetrometer 1 battery charge current	0.2	1%
7. Penetrometer 1 start signal and backup	0.2	±2 sec
8. Penetrometer 1 deployment signal and backup	0.2	±2 sec
9. Penetrometer 1 separation	0.2	±2 sec
10. Penetrometer 2 battery charge current	0.2	1%
11. Penetrometer 2 start signal and backup	0.2	±2 sec
12. Penetrometer 2 deployment signal and backup	0.2	±2 sec
13. Penetrometer 2 separation	0.2	±2 sec
14. Penetrometer 3 battery charge current		1%
15. Penetrometer 3 start signal and backup		±2 sec
16. Penetrometer 3 deployment signal and backup		±2 sec
17. Penetrometer 3 separation		±2 sec
18. Penetrometer 4 battery charge current		1%
19. Penetrometer 4 start signal		±2 sec
20. Penetrometer 4 deployment signal		±2 sec
21. Penetrometer 4 separation		±2 sec
22. Altimeter current		1%
23. Altimeter voltage		
24. Doppler current		
25. Doppler voltage		1%
26. Camera A shutter confirmation		±0.5 sec
27. Camera A beam current		1%
28. Camera A target voltage		
29. Camera A AGC voltage		
30. Camera A clamp (black) level		
31. Camera A high voltage		1%
32. Camera A erase lamp status		±1 sec
33. Camera B shutter confirmation		±0.5 sec
34. Camera B beam current		1%
35. Camera B target voltage		
36. Camera B AGC voltage		
37. Camera B clamp (black) level		
38. Camera B high voltage	0.2	1%

TABLE XXI (Cont'd)

DIAGNOSTIC DATA REQUIREMENTS LIST

Measurement	Sample Range	Accuracy
39. Camera B ease lamp status	0.2 sps	+1 sec
40. Camera C shutter confirmation		±.5 sec
41. Camera C beam current		1%
42. Camera C target voltage		
43. Camera C AGC voltage		
44. Camera C clamp (black) level		
45. Camera C high voltage		1%
46. Camera C erase lamp status		±1%
47. TV package temperature internal		5%
48. TV heater control voltage		5%
49. TV package pressure	0.2 sps	5%
50. Gimbal release confirmation		± 1 sec
51. Enable time gate pulse		±.5 sec
52. Photocell 1 current		1%
53. Photocell 2 current		
54. Photocell 3 current		
55. Photocell 4 current		1%
56. Camera A take picture command		±.5 sec
57. Camera B take picture command		±.5 sec
58. Camera C take picture command		±.5 sec
59. Sterilization canister pressure		5%
60. Depressurize sterile canister signal (prime)		± 1 sec
61. Depressurize sterile canister signal (backup)		
62. Depressurization valve operation		
63. Sterile canister separation signal (prime)		
64. Sterile canister separation signal (backup)		
65. Sterile canister MDF detonator, continuity loop monitor		
66. Sterile canister separation 1		
67. Sterile canister separation 2		
68. Sterile canister separation 3		
69. Sterile canister separation 4		
70. FS/FC mechanical separation signal (prime)		
71. FS/FC mechanical separation signal (backup)		
72. FS/FC separation		
73. FS/FC electrical separation signal (prime)		
74. FS/FC electrical separation signal (backup)		
75. Umbilical separation	0.2 sps	±1 sec

TABLE XXI (Cont'd)

DIAGNOSTIC DATA REQUIREMENTS LIST

Measurement	Sample Rate	Accuracy
76. ACS system current	0.2	2%
77. ACS system voltage		
78. ACS tank 1 pressure		
79. ACS tank 2 pressure		2%
80. ACS inverter frequency		0.1%
81. ACS inverter voltage phase 1		1%
82. ACS inverter voltage phase 2		1%
83. ACS inverter voltage phase 3		1%
84. ACS roll gyro output		
85. ACS pitch gyro output		
86. ACS yaw gyro output		
87. ACS roll error signal		
88. ACS pitch error signal		
89. ACS yaw error signal		1%
90. Gyro package temperature 1		1%
91. Gyro package temperature 2		1%
92. Gyro package temperature 3		1%
93. Roll reference angle		0.1%
94. Pitch reference angle		0.1%
95. Roll angle limit cycle verification		1%
96. Pitch angle limit cycle verification		1%
97. + Roll nozzle 1 operation		
98. + Roll nozzle 2 operation		
99. - Roll nozzle 1 operation		
100. - Roll nozzle 2 operation		
101. + Pitch nozzle 1 operation		
102. + Pitch nozzle 2 operation		
103. - Pitch nozzle 1 operation		
104. - Pitch nozzle 2 operation		
105. + Yaw nozzle 1 operation		
106. + Yaw nozzle 2 operation		
107. - Yaw nozzle 1 operation		
108. - Yaw nozzle 2 operation		1%
109. Gimbal angle 1		0.1%
110. Gimbal angle 2		
111. Gimbal angle 3		
112. Gimbal angle 4		0.1%
113. ACS manifold pressure 1		1%
114. ACS manifold pressure 2		1%
115. Roll rate inhibit monitor		5%
116. Pitch rate inhibit monitor		5%
117. Yaw rate inhibit monitor		5%

TABLE XXI (Cont'd)

DIAGNOSTIC DATA REQUIREMENTS LIST

Measurement	Sample Rate	Accuracy
118. ACS gas supply valve 1 monitor	0.2 sps	± 1 sec
119. ACS gas supply valve 2 monitor		± 1 sec
120. Propulsion ignition signal (prime)		± 0.1 sec
121. Propulsion ignition signal (backup)		± 0.1 sec
122. Propulsion igniter continuity loop		± 0.1 sec
123. Propulsion engine temperature		5%
124. Nose cap eject signal (prime)		+1 sec
125. Nose cap		
126. Nose cap separation		
127. Pressure port open signal (prime)		
128. Pressure port open signal (backup)		+1 sec
129. Parachute deployment signal (prime)		± 0.5 sec
130. Parachute deployment signal (backup)	0.2 sps	± 0.5 sec
131. Parachute mortar continuity loop		± 0.5 sec
132. Parachute deployment monitor		± 0.5 sec
133. Parachute load cell output		± 1%
134. Entry shell electrical disconnect signal (prime)		± 1 sec
135. Entry shell electrical disconnect signal (backup)		
136. Entry shell separation signal (prime)		
137. Entry shell separation signal (backup)		
138. Entry shell continuity monitor		
139. Entry shell separation		
140. Spare		
141. Spare		
142. Spare		
143. Spare		
144. Spare		
145. Spare		
147. Spare		
148. Change altimeter frequency command (prime)		
149. Change altimeter frequency command (backup)		
150. Change data mode command (prime)		
151. Change data mode command (backup)		± 1 sec
152. Transmitter 1 temperature		5%
153. Transmitter 1 Current		5%
154. Transmitter 1 Forward power		1%
155. Transmitter 1 Reverse power		1%
156. Transmitter 2 Temperature		5%
157. Transmitter 2 Current		5%
158. Transmitter 2 Forward power		1%
159. Transmitter 2 Reverse power		1%
160. Battery 1 charge current		1%

TABLE XXI (Concl'd)

DIAGNOSTIC DATA REQUIREMENTS LIST

Measurement	Sample Rate	Accuracy
161. Battery 1 temperature	0.2 sps	5%
162. Battery 1 voltage		1%
163. Battery 2 charge current		1%
164. Battery 2 temperature		5%
165. Battery 2 voltage		1%
166. External voltage 1 monitor		1%
167. External current 1 monitor		1%
168. External voltage 2 monitor		1%
169. External current 2 monitor	0.2 sps	1%
170. Sterile canister temperature 1	0.2 sps	5%
171. Sterile canister temperature 2		
172. Sterile canister temperature 3		
173-189. Heat shield temperature 1 through 16		
190. Afterbody internal pressure		
191.-198. Afterbody temperature 1 through 8	0.2 sps	5%
199. Parachute pack pressure		
200. Vibration X, Y and Z outer ring at 0 degrees		
201. Vibration X, Y and Z outer ring at 90 degrees		
202. Vibration X, Y and Z forward truss member 0 degrees		
203. Vibration X, Y and Z forward truss member 90 degrees		
204. Vibration X, Y and Z main support structure		
205.-220. Structure temperature 1 through 16		
221.-228. Heat shield ablation 1 through 8		
229.-236. Pressure 1 through 8		
237. Penetrometer 1 temperature		
238. Penetrometer 2 temperature		
239. Penetrometer 3 temperature		
240. Penetrometer 4 temperature		
241. TV package temperature (external) 1		
242. TV package temperature (external)2		
243.-254. Component box temperature 1 through 12		
255. ACS nozzle temperature 1		
256. ACS nozzle temperature 2		
257. ACS nozzle temperature 3	0.2 sps	5%

6.11.3 Functional Interfaces

Functional interfaces existing between the telecommunications subsystem and other functional areas in both the Flight Capsule and Flight Spacecraft are listed below according to functional area:

6.11.3.1 Electric Power and Control

The electric power and control subsystem shall provide nominal 28 vdc power as required by the telecommunication subsystem.

6.11.3.2 Programming and Sequencing

The programming and sequencing subsystem shall control the operation of the telecommunication subsystem.

6.11.3.3 Descent Retardation - Entry

The Flight Capsule telemetry antennas are to be located on the Flight Capsule afterbody as shown in Figure 8. Every consideration shall be given to removing non-dielectric objects from the field of view of the Flight Capsule antennas. The Flight Spacecraft relay antenna shall be located on the Flight Spacecraft such that the peak radiation vector occurs at a clock angle of 282 degrees and a cone angle of 110 degrees.

6.11.3.4 Thermal Control

The telecommunication subsystem temperature shall be maintained within temperature extremes of -20°C and +80°C operating, and -55 and + 135°C nonoperating, with the exception of the Flight Spacecraft portion of the telecommunication subsystem. The Flight Spacecraft portion of the telecommunication subsystem shall be maintained within the same operating temperatures as the Flight Spacecraft, but the nonoperating extremes shall be -55 and +100°C.

6.11.3.5 Descent Retardation - Post-Entry

Materials used in the parachute subsystem shall be dielectric wherever possible to minimize their effects on the telemetry antenna performance.

6.11.4 Component List

The following listed components represent the major components in the telecommunications subsystem:

1. FC radio subsystem

2. FC data handling subsystem

3. FS relay radio subsystem

4. FS relay data handling

PART II

COMPONENT SPECIFICATIONS

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INTRODUCTION

The primary functional area requirements presented in Part I generally specify an adequate level of detail from which various solutions to each subsystem required could be developed for an integrated Capsule System definition. However, to select the Flight Capsule design, related to the Capsule System and the rest of the systems in the Project, required a unique solution of many subsystem areas. Therefore, component specifications were prepared for those subsystem areas which required more than a catalog level of technical definition.

Further design evolution of the Flight Capsule will result in greater definition and/or changes to the component specification details plus additional component specifications. In some cases the design solution presented is one of several which could accomplish the intended function of the subsystem. As a result it is not intended that the enclosed specifications express the only techniques or approach to satisfy the primary functional area requirement, but represent one way such requirements could be met.

An asterisk following any paragraph in these component specifications indicates the information was not available at the time these specifications were prepared or that the information was not critical to the Flight Capsule design synthesis.

STERILIZATION CANISTER COMPONENT SPECIFICATION

Number III-2-2-A

1.0 SCOPE

This document specifies the performance and design requirements of the sterilization canister as a subsystem of the Flight Capsule.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The sterilization canister shall consist of the following parts.

3.1.1 Shell

The shell shall consist of a lid, a two piece base, and a rear door in accordance with drawing K4-1544, Flight Capsule Launch Configuration.

3.1.2 Separation Mechanism

The separation mechanisms shall consist of an elastomer encased mild detonating fuze (MDF), a backup ring, a support ring, and a shield in accordance with drawing K4-1544.

3.1.3 Pressure Control Assembly

The pressure control assembly shall consist of the following parts in accordance with drawing K4-1544.

3.1.3.1 Tank

A pressure tank to store the gas used to replace the gas lost through leakage from the canister.

3.1.3.2 Regulator

A regulator to sense the differential between that pressure inside the canister and the ambient pressure; and to bleed the tank pressure to replace gas lost through canister leakage.

3.1.3.3 Relief Valve

A relief valve to bleed over pressure from the inside of the canister during the launch phase of the mission.

3.1.3.4 Fill Valve

A valve to allow pressure supply and regulation from outside during storage after sterilization to launch.

3.1.3.5 Depressurization Valve

A valve to allow closing of the tank bleed line and opening of the canister to allow depressurization.

3.1.3.6 Filters

Microfilters in all lines leading to the outside of the canister to eliminate the possibility of contamination of the entry vehicle.

3.2 Operation

The sterilization canister assembly shall enclose the Entry Vehicle and act as a passive thermal control and sterilization barrier. As a thermal control barrier, it shall reduce the radiation of heat from the Entry Vehicle during planetary transfer. As a sterilization barrier it shall prevent the contamination of the Entry Vehicle from the sterilization heat cycle to opening of the lid prior to separation of the Entry Vehicle from the Flight Spacecraft.

To prevent recontamination, the sterilization canister assembly shall include a pressurization tank and controls to maintain an internal pressure greater than ambient. It shall also include a valve to depressurize the canister at a selected time and a separation mechanism to jettison the canister lid prior to deployment of the Entry Vehicle.

3.3 Characteristics

The sterilization canister assembly shall conform to all requirements of drawing K4-1544.

3.3.2 Mass Properties

3.3.2.1 Weight (total assembly): 323 pounds maximum

3.3.2.2 Center of Gravity: to be determined

3.3.2.3 Moments of Inertia

I_{zz} (pitch)	to be determined
I_{yy} (yaw)	to be determined
I_{xx} (roll)	to be determined

3.3.2.4 Products of Inertial

I_{xy}	to be determined
I_{yz}	to be determined
I_{yz}	to be determined

3.3.3 Environments

The sterilization canister assembly shall function after exposure to the environments specified in paragraph 4.3 of applicable document 2.1.

3.3.4 Leakage

Sample tests shall demonstrate leakage through seals, welds are not to exceed 27.5×10^{-8} lb/sec of nitrogen from a completely assembled canister with an external pressure of 10^{-6} microns and an internal pressure of 1.0 lb/in^2 psi. Test methods to be approved by contracting agency.

3.3.5 Separation

Sample tests shall demonstrate a clean separation with a minimum separation impulse of 5.59 lb/sec. Test methods to be approved by contracting agency.

3.3.6 Pressure Regulation

1. Pressure regulation shall be maintained from sterilization to booster mating by an external source through the fill valve. From booster mating to depressurization, pressure regulation shall be maintained by the canister pressurization subassembly.

2. Canister internal pressure shall be regulated to 1.0 psi above ambient.

3. The depressurization valve actuation shall close the tank outlet and reduce the canister pressure to 1.0 psia within a maximum of 5 minutes.

3.3.7 Thermal Control Surface Emissivity - to be determined.

3.4 Interfaces

3.4.1 Electrical

3.4.1.1 Separation connector - two hermetically sealed 60 pin bulkhead connectors shall pass through the canister shell. No inflight separation is required.

3.4.1.2 Control Signals Required

1. To shut-off valve

2. To depressurization valve

3. To lid separation ignitor circuit.

3.4.1.3 Signal Sources

1. Shut off valve - from ground handling equipment

2. Depressurization valve and lid separation circuit - from Flight Capsule CC & S

3.4.1.4 Initiating Current - 4.5 amp dc minimum

3.4.2 Mechanical

The fill valve opening shall mate with an auxiliary pressure source from sterilization to launch. The FC to FS adapter forward and aft sections shall attach to the canister base and carry booster thrust loads to the Entry Vehicle.

3.4.3 Thermal

To be determined

CENTRAL COMPUTER AND SEQUENCER COMPONENT SPECIFICATION

Number III-2-2-B

1.0 SCOPE

This document specifies the performance and design requirements of the central computer and sequencer as a component of the Flight Capsule programming and sequencing subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The central computer and sequencer (CC &S) shall consist of a single assembly containing the following functional parts:

1. Master clock and sequencer
2. Electrical stimulation sequencer
3. Separation timer
4. Backup rocket ignition timer
5. Mach number computer
6. Entry shell ejection computer
7. TV camera control
8. Main chute timer
9. Penetrometer timer
10. Quantitative command storage

3.2 Operation

The central computer and sequencer shall, in response to one or more discrete commands, (1) initiate system checkout and calibration, (2) initiate attitude control and rocket ignition, (3) initiate parachute deployment, (4) initiate entry shell, nose cap and penetrometer jettison, and (5) initiate television picture taking.

3.3 Characteristics

3.3.1 Master Clock and Sequencer

The master clock shall supply a clock pulse of between 1 kilocycles and 100 kilocycles to the television system. The master clock and sequncer shall be capable of initiating up to 52 events at 20 different times over a 30 hour period. Prior to final assembly, 12 of the times may be set. The remaining eight times shall be stored in the quantitative command storage and shall be subject to change up until a few minutes prior to separation. The clock shall have a 1 minute resolution. The events shall be as follows:

Number of Events	Form of Event
3	Turn on internal clocks in CC & S
8	Digital pulse train to ACS
7	Operate relay - 500 mw, 28 vdc 0.5 second pulse
34	Operate semiconductor switch in power control 300 mw at 3 vdc, 50 μ/sec pulse.

The master sequencer shall be initiated by DC 1. Discrete command (DC) 6 shall initiate the checkout sequence. The checkout sequence consists of the first 11 events of the master sequence. On completing these 11 events, the CC & S shall return all switches to the position that they were in prior to initiation of the sequence. The initiation of the fourth time dependent event of the checkout sequence shall be accomplished only if DC 7 accompanies DC 6.

3.3.2 Electrical Stimulation Sequencer

The electrical stimulation sequencer (ESS) shall be initiated by the master sequencer.

3.3.2.1 Accelerometer Calibration

The ESS shall supply a 0, 25, 50, 75 and 95 percent calibration signal. Each signal shall be supplied for 10 seconds, followed by a step increase to the next signal for 10 seconds. After the 95 percent signal has been applied for 10 seconds the signal shall decrease in a linear manner in 10 seconds to zero percent.

Accelerometer calibration signals shall be provided at six pairs of terminals simultaneously. The exact nature of the signals (voltage, current, frequency) shall be specified at a later date.

3.3.2.2 IRS/ACS/TV Platform Stimulation

The ESS shall supply currents to each of six pairs of terminals in sequence. The currents shall be applied for 10 seconds each. The currents shall be specified at a later date.

3.3.2.3 Altimeter Calibration

The ESS shall supply 500 mv at 28 vdc to one pair of terminals for 50 seconds and shall supply 500 mv at 28 vdc to a second pair for 50 seconds.

3.3.3 Separation Timer

The separation timer shall be initiated by the eighth timed event of the master sequence. The separation timer shall supply three pulses. The second pulse shall follow the first by 1 second, the third pulse shall follow the second by 5 seconds.

This sequence shall be inhibited until DC 3, and a separation switch opening have occurred.

The first and third pulses shall be 500 mw a 28 vdc for 0.5 second. The third pulse shall be 300 mw at 3 vdc for 50 μ/sec.

3.3.4 Backup Rocket Ignition Timer

The backup rocket ignition timer shall be intiated by closure of a switch for 3 minutes, it shall supply two 300 mw at 3 vdc for 50 μ/sec pulses.

3.3.5 Mach Number Computer

The Mach number computer shall receive input signals from the accelerometer and altimeter. The accelerometer output shall be square wave whose frequency is proportional to instananeous acceleration. The computer shall count the frequency every half second. When three successive counts are lower than the previous count, the time shall be 2 seconds after peak entry acceleration (g max). The computer shall then solve for the following times:

$$t_4 = (K_1 \text{ g max.})^{-A}$$

$$t_5 = (K_2 \text{ g max})^{-B}$$

$$t_7 = (K_3 \text{ g max.})^{-C}$$

K_1, K_2, K_3, A, B and C are all stored quantitative commands (QC). Altitudes F, G and H are stored quantitative commands. The Mach number computer shall supply two 300 mw, 3 vdc, 50 μ/sec pulses to start the TV camera control and to start the main parachute timer when time is equal to or greater than t_4 and altitude is less than F, or time is equal to or greater than t_5, or altitude is less than G. The Mach number computer shall start the penetrometer timer and transmit a command to change data mode when time from maximum g is equal to or less than t_7 or when altitude is less than H.

3.3.6 Entry Shell Ejection Computer

The entry shell ejection computer shall receive inputs from the accelerometer and parachute load switch. When the parachute load switch closes or when the accelerometer output indicates a 5g increase in deceleration in 0.5 second, the entry shell ejection computer shall supply four 300 mw, 3 vdc, 50 μ/second pulses and two 500 mw, 28 vdc, 0.5 second pulses. During checkout, this computer shall respond to the simulated change in acceleration produced by the electrical stimulation sequencer.

3.3.7 TV Camera Control

The TV camera control shall be initiated by the pilot parachute deployment command or by closure of a separation switch.

3.3.7.1 The camera control shall supply a pulse to take a set of pictures if the following conditions are satisfied:

1. Five seconds have elapsed since the pilot parachute command was given or 2 seconds have elapsed since closure of the separation switch,

2. Camera platform limit switches are open,

3. Voltage is not present on $|\theta_c|$ or $|\dot{\theta}_c|$ terminals,

4. 8.2 seconds have elapsed since item 1 occurred and items 2 and 3 are not yet satisfied.

3.3.7.2 The camera control shall supply a signal 6 seconds after the "take-picture" pulse. This pulse shall command the engineering data handling to start reading data out of memory.

3.3.7.3 Twelve seconds after the receipt of a pulse (shutter confirmation), the camera control shall supply a second pulse to take picutres if conditions 2 and 3 or condition 4 above are satisfied. 3.3.7.3 shall be repeated continuously until the end of the mission.

3.3.8 Main Parachute Timer

The main parachute timer shall be initiated by the Mach number computer. Two seconds after initiation, the main parachute timer shall supply two 300 mw, 3 vdc, 50 μ second pulses. Five seconds after initiation the timer shall supply two 300 mw, 3 vdc, 500 μ second pulses.

3.3.9 Penetrometer Timer

The penetrometer timer shall be initiated by the Mach number computer. It shall supply a pair of 300 mw, 3 vdc, 50 μ second pulses to a different pair of terminals every 2 seconds for a total of 4 pairs of pulses of 300 mw at 3 vdc.

3.3.10 Quantitative Command Storage

The QC storage shall receive, store, and provide, on interrogation, 23 quantitative commands as listed in paragraph 3.4.1.2.

3.4 Interfaces

3.4.1 Electrical Interfaces

3.4.1.1 Inputs to CC & S

The following discrete commands (DC) from the flight spacecraft shall be received and appropriately processed:

DC 1 Start master sequence

DC 2 Inhibit override - stable platform dirft

DC 3 Separation enable

DC 4 Spare

DC 5 Inhibit override - sterilization canister pressure

DC 6 Start checkout sequence

DC 7 Release calibration gas sample

DC 8 Arm separation subsystem

DC 9 Start electrical stimulation sequence

DC 10 Open cold gas supply valve

DC 11 Separate FC from FS

DC 12 Inhibit Override - sterile canister lid separation switch

DC 13 Inhibit Override - reaction control pressure.

3.4.1.2 The following quantitative commands from the flight spacecraft shall be stored in the CC & S quantitative command storage:

QC 1 Thrust vector roll command

QC 2 Thrust vector pitch command

QC 3 Spare

QC 4 Spare

QC 5 Time of first cruise maneuver sequence

QC 6 First maneuver roll angle

QC 7 First maneuver pitch angle

QC 8 Time of second maneuver sequence

QC 9 Second maneuver roll angle

QC 10 Second maneuver pitch angle

QC 11 Time of third maneuver sequence

QC 12 Third maneuver roll angle

QC 13 Third maneuver pitch angle

QC 14 Predicted time of entry

QC 15 Constant K_1 for maximum parachute deployment Mach number

QC 16 Constant A for maximum parachute deployment Mach number

QC 17 Constant K_2 for minimum parachute deployment Mach number

QC 18 Constant B for minimum parachute deployment Mach number

QC 19 Constant K_3 for penetrometer deployment Mach number

QC 20 Constant C for penetrometer deployment Mach number

QC 21 Altitude F maximum parachute deployment altitude

QC 22 Altitude G minimum parachute deployment altitude

QC 23 Altitude H penetrometer deployment altitude

3.4.1.3 The following miscellaneous commands from the sources noted shall be appropriately processed:

	Source	Form of Signal
1.	Reaction control manifold pressure	$P>15$ psia/$P<15$ psia
2.	FC to FS separation	open/closed
3.	Sterilization canister lid separation	open/closed
4.	Acceleration	Square wave-frequency proportional to acceleration
5.	Parachute riser line load	open/closed
6.	Pressure-sterilization canister	on/off
7.	Nose cap separation	on/off
8.	Altitude - above 27,500 feet	Digital pulse train
9.	Altitude - below 27,500 feet	Digital pulse train
10.	Camera platform, limit switch	on/off
11.	Altitude confirmation	on/off

3.4.1.4 A maximum of 6 watts at 28 vdc (nominal) shall be supplied from the electrical power and control subsystem

3.4.1.5 Outputs From CC & S

1. Master Clock and Sequencer:

   1 clock frequency
   3 internal signals
   8 digital pulse trains
   7 signals, 500 mw, 28 vdc
   34 signals, 300 mw, 3 vdc

2. Electrical Stimulation Sequencer:

   5 levels of accelerometer calibration signal
   6 currents
   2 voltages

3. Separation Timer:

   2 pulses, 500 mw, 28 vdc
   1 pulse, 300 mw, 3 vdc

4. Backup Rocket Ignition Timer:

   2 pulses, 300 mw, 3 vdc
   1 command

5. Mach Number Computer:

   2 pulses, 300 mw, 3 vdc
   1 command

6. Entry Shell Ejection:

   4 pulses, 300 mw, 3 vdc
   2 pulses, 500 mw, 28 vdc

7. TV Camera Sequences:

   Series of pulses

8. Main Parachute Timer:

   4 pulses, 300 mw, 3 vdc

3.4.2 Mechanical Interfaces

1. Weight: 8 pounds
2. Volume: 200 $in.^3$

3.4.3 Thermal Interface

Electrical Power, insulation and surface coatings to maintain the temperature between -20 to +80°C operating, and -55 to +135°C storage.

ROCKET MOTOR COMPONENT SPECIFICATION

Number III-2-2-C

1.0 SCOPE

The document specifies the performance and design requirements of the rocket motor as a component of the Flight Capsule propulsion subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

2.2 Exhaust Nozzle Extension Component Specification, Number III-2-2-D

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The rocket shall supply a change in velocity to the Separated Vehicle.

3.2 Operation

3.2.1 Total Impulse, lb-sec. --- 101,600 ± 1 percent, 3 sigma

3.2.2 Specific Impulse, second (vacuum) --- 254

3.2.3 Thrust Nominal, pound -- 3000

3.2.4 Temperature Environment - Exclusive of sterilization:

Storage: - 40 to 175°F
Operating: - 40 to 175°F

3.2.5 Space Storage -- Up to one year at 10^{-6} torr

3.3 Characteristics

3.3.1 Propellant Mass Fraction --- 0.93

3.3.2 Propellant - Solid TP-H-3105

3.3.3 Type Motor

Spherical with submerged exhaust nozzle, entire assembly to be sterilizable at both component and assembled level.

3.3.4 Motor Size

1. Diameter Outside, inches -- 22.3
2. Length, inches 24.0
3. Exit Area, square inches 74.93
4. Expansion Ratio 18.7

3.3.5 Case Design -- 6Al, 4V Titanium 0.026 inch thick

3.3.6 Weight

1. Propellant, pounds 400.0
2. Inert parts, pounds 28.7
3. Pyrogens, pounds 2.7
4. Liner, pounds 0.6
5. Total, pounds 432.0

3.3.7 Igniter Location

The igniter shall be installed at nozzle end of the motor case.

3.3.8 Thrust Vector Alignment

Thrust vector alignment shall be within 0.5 degree (3 sigma) cone angle around the theoretical thrust axis.

3.4 Interfaces

3.4.1 Electrical

An ignitor initiating current of 4.5 amperes shall be supplied.

3.4.2 Mechanical

3.4.2.1 Rocket Motor Mounting

A mounting flange shall be provided with a bolt hole circle of 23.25 inches diameter. The flange plane shall be perpendicular to motor thrust axis within 0.1 degree, located 14.5 inches from the motor head end.

3.4.2.2 Exhaust Nozzle Extension

A mounting shall be provided for an exhaust nozzle extension to be attached to the motor nozzle exit plane.

EXHAUST NOZZLE EXTENSION COMPONENT SPECIFICATION

Number III-2-2-D

1.0 SCOPE

This document specifies the performance and design requirements of the exhaust nozzle extension as a component of Flight Capsule propulsion subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part 1

2.2 Rocket Motor Component Specification, Number III-2-2-C

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The exhaust nozzle extension shall duct the rocket motor exhaust gases away from the Separated Vehicle structure.

3.2 Operation

3.2.1 Temperature Environment -- Exclusive of sterilization:

Storage: -40 to 175°F

Operating: -40 to 175°F

3.2.2 Operating Temperature

The exhaust nozzle extension shall be exposed to the exhaust products of the rocket motor.

3.3 Characteristics

3.3.1 Material

The material shall be dielectric prior to, during and after exposure to exhaust products of the rocket motor.

3.3.2 Structure

Fiberglass structure 0.050 inch thick coated with Teflon, 0.135 inch and 0.080 inch on the inside and outside surfaces respectively.

3.3.3 Weight

9 pounds maximum

3.3.4 Size

11 inches long with an internal diameter of 9.77 inches at nozzle attachment plane and 16.5 inches at exit plane.

3.3.5 Shape

Continue contour of rocket motor exhaust nozzle.

3.4 Interface

3.4.1 Mechanical

Provide mounting flange to be compatible with rocket motor exhaust nozzle exit plane.

INERTIAL REFERENCE SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-E

1.0 SCOPE

This document specifies the performance and design requirements of the inertial reference subsystem as a component of the Flight Capsule attitude control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Cold Gas Reaction Control Subsystem, Component Specification Number III-2-2-G

2.3 Hot Gas Thrust Vector Control Subsystem, Component Specification, Number III-2-2-F.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The inertial reference subsystem (IRS) shall include the following major assemblies:

1. Four-Gimbal Inertial Platform
2. Digital Computer
3. Rate Sensor

The inertial platform shall provide a method of establishing and maintaining an attitude reference for the entry vehicle. The digital computer shall transform the information generated by the platform (gimbal angles, accelerometer data) into the proper reference frame for controlling the vehicle attitude and other required functions. The rate sensor assembly shall measure body angular rates and shall disable the attitude control reaction system in the event that the rates exceed a predetermined limit.

3.2 Operation

The IRS shall be deactivated from launch until just prior to separation of the Flight Capsule from the Flight Spacecraft. The IRS shall be energized at least 1 hour prior to any checkout procedure to allow sufficient stabilization of the inertial sensors.

Checkout procedures shall exercise the IRS and determine its operating characteristics (drift, command receiving ability, controlling signal characteristics, etc.). Checkout procedures shall determine a go or no-go condition.

After completing the checkout , the IRS shall be energized in a ready position for separation. The platform shall be caged to the Separated Vehicle axis. The command angles to the IRS for attitude control shall be zero at this time. Just prior to separation (1 or 2 seconds) the inertial platform shall be uncaged and be in an inertial mode thus providing an inertial reference by virtue of the platform gyros. The inertial platform and digital computer shall provide signals to a cold gas reaction subsystem to maintain the vehicle attitude. The digital computer will receive commanded changes in the attitude from the Flight Capsule central computer and sequencer (CC&S). Any number of these commands can be accepted by the digital computer from separation until mission completion and the IRS shall provide the controlling signals to the reaction control subsystem to achieve and maintain the commanded attitude.

On command from the CC&S, the rate sensor system shall be activated to provide a signal to the IRS computer. If the body rates exceed a predetermined value (approximately 6 deg/sec) a signal from the computer will deactivate the ACS reaction control system.

The IRS shall provide accelerometer data to the CC&S for various Flight Capsule functions and to the engineering data handling subsystem for data purposes. On command from the CC&S the IRS shall discontinue providing attitude control signals to the reaction control subsystem and substitute roll rate limiting signals. The inertial reference shall be maintained until impact. The CC&S shall issue a command to the IRS to provide gimbal angle commands to an auxiliary gimbal system for controlling the TV camera platform. These control signals shall be in analog form for positioning and maintaining the auxiliary gimbal system to a predetermined angular position with respect to the inertial frame being maintained by the IRS platform.

3.3 Characteristics

3.3.1 Performance

The IRS shall have the following performance capabilities for maintaining an attitude reference:

1. G-Insensitive Drift: 0.4 deg/hr (1σ)
2. G-Sensitive Drift: 0.3 deg/hr/g (1σ)
3. Readout Error: 0.1 degree (1σ)
4. Computation Error: 0.1 degree (1σ)
5. Maximum Rate Capability: 1000 deg/sec

The rate sensor assembly shall provide a signal proportional to the angular rate from 0 to 20 deg/sec with an accuracy of 5 percent. Accelerometers shall be mounted on the inner gimbal of the platform for engineering data purposes.

3.3.2 Weights and Envelopes

Assembly	Weight (pounds)	Volume (cubic inches)	Power (watts)
Inertial Platform	10	400	45
Computer	5	160	10
Rate Gyro	2	50	13
Cables	1	-	-
total	18	610	68

3.4 Interfaces

Power shall be provided by the power control subsystem and shall be 28 vdc ± 1 percent. The commands to the IRS shall be provided by the CC&S and the IRS will provide control signals (28 vdc) to the reaction control valves. The IRS shall provide signals to a camera gimbal for controlling its angular pointing. Accelerometer data shall be provided to the CC&S from the inner gimbal.

HOT GAS THRUST VECTOR CONTROL SUBSYSTEM
COMPONENT SPECIFICATION

Number III-2-2-F

1.0 SCOPE

This document specifies the performance and design requirements of the hot gas thrust vector control subsystem as a component of the Flight Capsule attitude control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

2.2 Inertial Reference Subsystem, Component Specification, Number III-2-2-E

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

3.1.1 Components

The hot gas thrust vector control subsystem (TVC) shall include the components illustrated in Figure 1. Each of the four thrust modules shall consist of a solid propellant gas generator and two opposed normally open nozzle solenoid valves.

3.1.2 Vehicle Installation

The four thrust modules shall be located with the vehicle structure so as to place four nozzle thrust axes in the plane of the vehicle pitch and yaw axes, respectively. The gas generators shall be located to effect the minimum possible length of tubing between the generator output port and valve inlet port.

3.2 Operation

3.2.1 System Actuation

The TVC system operation shall be started by application of the initiation voltage described in paragraph 3.4.1 to each of the gas generator igniters.

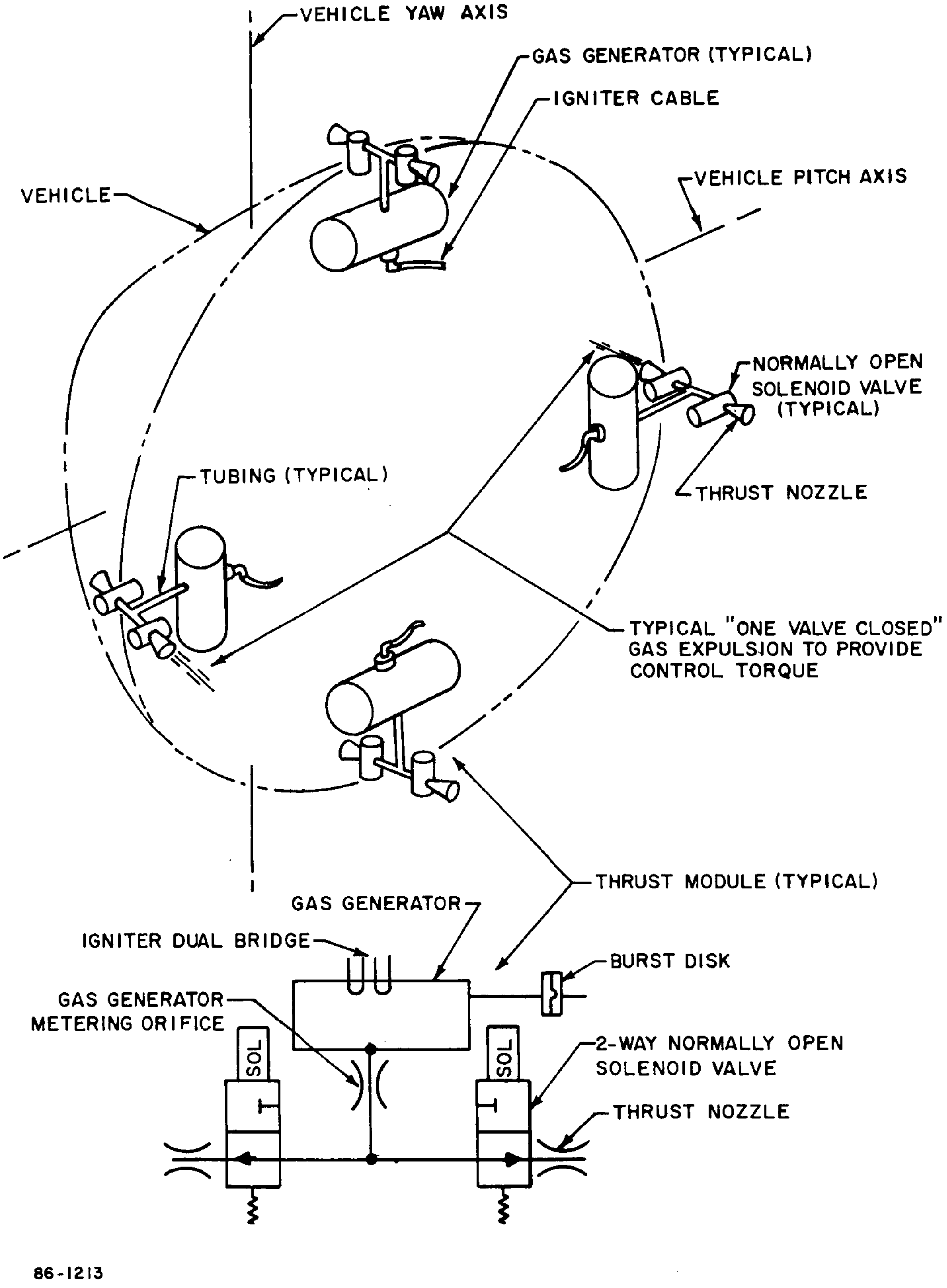


Figure 1 SOLID PROPELLANT OPEN CENTERED TVC SYSTEM

3.2.2 Gas Generator

Each gas generator shall provide a constant output weight flow rate of 0.115 lb/sec of noncorrosive gaseous products of combustion. The specified flow rate shall not be affected by closure of each nozzle solenoid valve and shall be present for a duration of 35 seconds after initiation. Accidental closure of both solenoid valves shall cause the opening of a pressure relief burst disk extinguishing combustion of the propellant and yielding a zero net thrust vector as a result of the malfunction. The generator combustion process shall not require a pressure greater than 2500 lb/in^2 psi nor a flame temperature in excess of 2000°F.

3.2.3 Thrust Modulation

Electrical actuation of opposed solenoid nozzle valves (closure of two valves) on two thrust modules shall cause 25 pounds of thrust to be applied by each of the remaining open valves. (Illustrated in Figure 1 as an example for the pitch axis.) Each solenoid nozzle valve shall possess dynamic response parameters not in excess of: time delay = 0.010 seconds and time constant = 0.005 second.

3.3 Characteristics

3.3.1 Component Weight

The weight breakdown of the system shall not exceed the values tabulated below:

Component	Unit Weight (pounds)	No. Req'd.	Total (pounds)
Gas generator	7.77	4	31.1
Solenoid nozzle valve	0.52	8	4.2
Tubing, brackets, etc.	--	--	5.0
		Max. Weight =	40.3

3.3.2 Component Envelopes

The size of the subsystem components shall not exceed the dimensions of the right circular cylinders described as follows:

(a) Gas Generator - 4.5 inch diameter x 6 inches in length

(b) Solenoid Nozzle Valve - 3.5 inches in diameter x 3 inches in length.

3.4 Interface

3.4.1 Electrical

The gas generator initiators shall be of the dual bridge wire type characterized by a no-fire current of 1 ampere or 1 watt for 5 minutes and an all fire current of 4.5 amperes both at a voltage level of 28 vdc. The initiator shall be capable of igniting the propellant within 0.050 second after application of the all fire current. The solenoid nozzle valves shall perform as specified in Section 3.2.3 with an excitation of 28 vdc and shall not require greater than 2 watts of power.

3.4.2 Mechanical

The exterior surface temperature of any component shall not exceed 1000°F after 35 seconds of subsystem operation.

COLD GAS REACTION CONTROL SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-G

1.0 SCOPE

This document specifies the performance and design requirements of the cold gas reaction control subsystem as a component of the Flight Capsule attitude control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Inertial Reference Subsystem, Component Specification, Number III-2-2-E.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

3.1.1 Components

The cold gas reaction control subsystem (RCS) shall include the components illustrated in Figure 1. The configuration shall employ identical redundant subsystems. Gaseous nitrogen propellant shall be stored under pressure in the pressure vessels by means of the fill and vent valves shown in series with filters and the vessels. Normally closed squib valves shall serve to contain the propellant until system activation. Each pressure vessel shall supply propellant to a pressure regulator and six normally-closed solenoid nozzle valves.

3.1.2 Vehicle Installation

The twelve solenoid nozzle valves shall be located within the vehicle structure so that selective valve actuation provides plus or minus torque couples about the three vehicle principal axes. Each of the redundant subsystems shall provide one half the torque couple about a given axis and all thrust vectors shall be located in the planes of the vehicle pitch, roll and yaw axes. The fill and vent valves shall be located so as to be accessible for attachment of ground support pressurization equipment. All seal joints shall be accessible for leak testing.

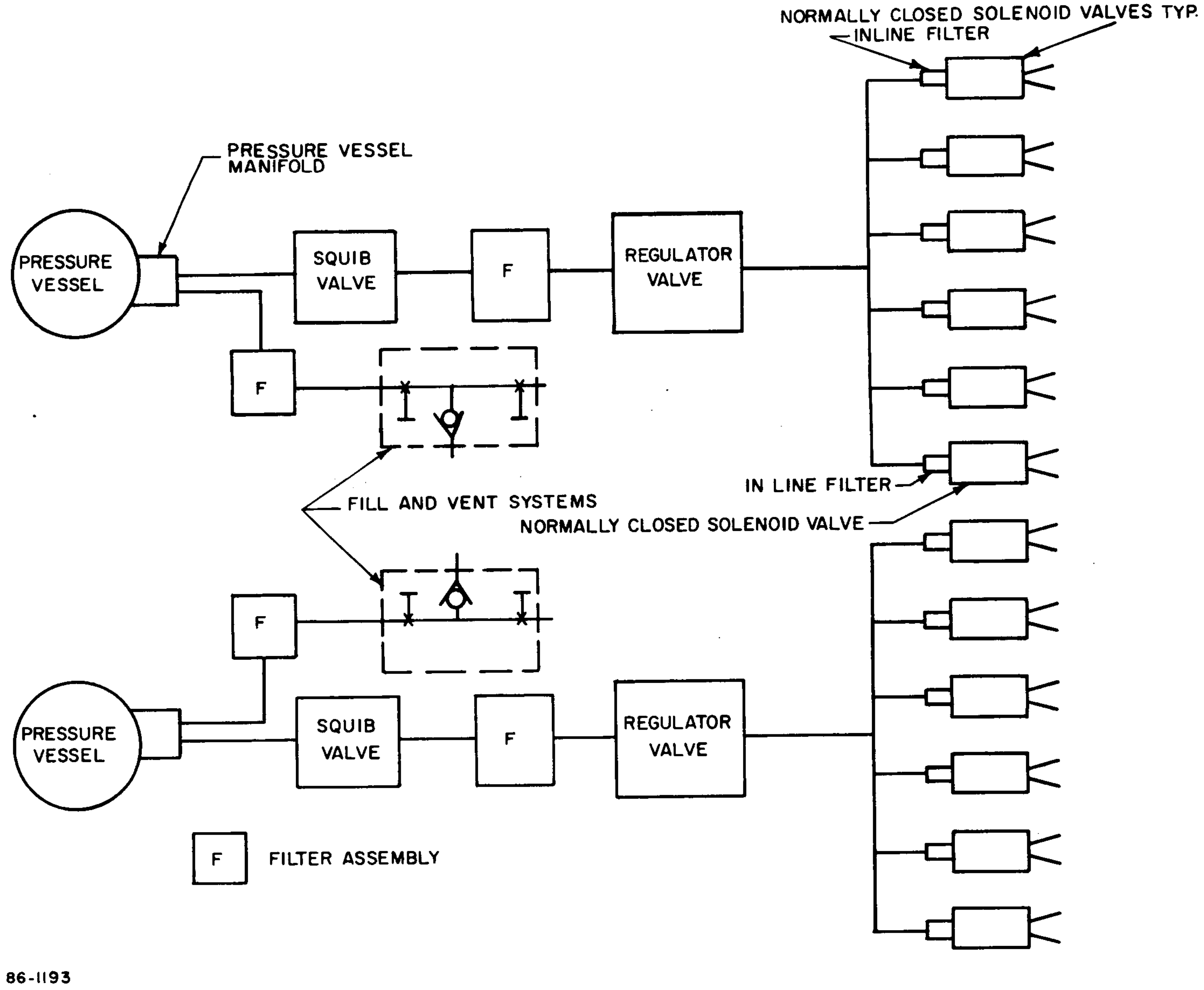


Figure 1 REACTION CONTROL SYSTEM SCHEMATIC LAYOUT

3.2 Operation

3.2.1 Subsystem Initiation

The reaction control subsystem operation shall be initiated by application of power described under paragraph 3.4.1 to both of the squib valve pyrotechnic igniters.

3.2.2 Pressure Vessels

Each of the pressure vessels shall deliver a total impulse of 124 lb-sec. minimum, utilizing a gaseous nitrogen propellant and an expulsion process yielding a propellant specific impulse of 60 seconds minimum. The duration of impulse delivery shall not exceed 2 hours.

3.2.3 Squib Valve, Filter, Regulator, and Tubing

The components connecting the pressure vessels to the solenoid nozzle valves shall operate as follows:

3.2.3.1 Squib Valve

Contain the nitrogen propellant in the pressure vessel until actuation. After actuation, provide a flow path for the propellant to the filter inlet.

3.2.3.2 Filter

Remove chips produced by the squib valve actuation as well as particulate contamination in general from the propellant.

3.2.3.3 Regulator

Reduce the pressure of the propellant at the solenoid nozzle valve inlets to a level compatible with the valve operating requirements described in section 3.2.4.

3.2.3.4 Tubing

Provide flow paths from the regulator outlet to each of the solenoid valve inlets.

Each of the above components shall operate such that the regulated pressure at the inlet of any nozzle valve shall not cause thrust deviation from the value specified in paragraph 3.2.4 regardless of which group of three nozzle valves are open simultaneously at any instant of time on either subsystem.

3.2.4 Solenoid Nozzle Value

Electrical actuation of a given solenoid nozzle valve shall cause the thrust value tabulated below to be applied to the vehicle.

Axis	Thrust value per nozzle	Thrust tolerance
Yaw	0.5 lbs	± 0.05 lbs
Pitch	0.5 lbs	± 0.05 lbs
Roll	0.5 lbs	± 0.05 lbs

Each solenoid nozzle valve shall possess dynamic response parameters not in excess of: a. time delay = 0.20 second b. time constant = 0.005 second.

3.3 Characteristics

3.3.1 Component Weight

The weight breakdown of the system shall not exceed the values tabulated below:

Component	Unit Weight (pounds)	No. Req'd.	Total (pounds)
Solenoid nozzle valve	0.18	12	2.2
Pressure vessel	5.90	2	11.8
Propellant (N_2)	5.00	--	5.0
Squib valve	0.37	2	0.7
Vessel manifold	0.25	2	0.5
Fill and vent valve	0.75	2	1.5
Tubing complex	2.85	2	5.7
Regulator	3.00	2	6.0
Filter	0.25	4	1.0
		Max Wt. =	34.4 lbs.

3.3.2 Component Envelopes

The size of the principal subsystem components is shown in figure 1.

3.4 Interface

3.4.1 Electrical

The squib valve initiators shall be of the dual bridge wire type characterized by a no-fire current of 1 ampere or 1 watt for 5 minutes and an all fire current of 4.5 amperes both at a voltage level of 28 vdc. The solenoid nozzle valves shall perform as specified in paragraph 3.2.4 with an excitation of 28 vdc and shall not require greater than 0.5 watt per valve.

3.4.2 Thermal

The cold gas reaction control subsystem shall be maintained at a temperature between -100 and +140°F (300°F max. during sterilization.)

TELEVISION SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-H

1.0 SCOPE

This document specifies the performance and design requirements of the television subsystem as a component of the Flight Capsule engineering data acquisition subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The television experiment shall provide high resolution images of the Martian surface during the parachute descent phase of the Flight Capsule mission. The television subsystem shall provide the following specific functions to accomplish the television experiment:

1. Camera shuttering (on external command)
2. Video amplification and automatic gain control (AGC) if required
3. Scan control and synchronization
4. Beam current regulation
5. Power conditioning and high voltage generation
6. Focus control
7. Blanking
8. Vidicon erasure and priming (if required)
9. Analog-to-digital conversion of video signals to 5-bit precision
10. Line and camera identification and performance indication
11. Camera stabilization
12. System checkout and calibration after installation in the Flight Capsule (on external command)

13. Suitable test points for diagnostic telemetry

14. Active thermal control and environmental protection as required.

The hardware required to implement these functions shall be arbitrarily divided into six functional elements:

1. Cameras

2. Power conditioning

3. Camera stabilization platform

4. Image programming

5. Data generation and processing

6. Environmental packaging

3.2 Operation

The subsystem shall take television pictures of the Martian surface during Flight Capsule descent on the parachute nominally from 27,500 feet to impact. The subsystem shall produce as many images as possible (but at least 9 images) consistent with the following major constraints:

1. All picture data shall be transmitted from the Flight Capsule to Flight Spacecraft prior to impact.

2. A nominal 15,000k bits/sec transmission rate shall be available for TV data.

3.3 Characteristics

3.3.1 Cameras

3.3.1.1 Resolution and Field

Three fixed focus, fixed focal length television cameras shall be required with nominal resolution and field of view as follows:

Camera	Field of view (degrees)	Resolution (for vertical image at 20,000 ft. attitude)
A-Camera	24	30 ft/TV line
B-Camera	8.2	10 ft/TV line
C-Camera	2.7	3.3 ft/TV line

At least one image is required at lower altitude to yield 1 ft/ TV line resolution. The cameras shall remain in focus from 30, 000 to 1, 000 feet.

3.3.1.2 Image Format

The television image format shall consist of 200 lines X 200 electrically resolvable picture elements. Each image element is quantized to at least 32 grey levels.

3.3.1.3 Nesting

Nested images covering a 9: 3: 1 resolution range are required. The required nesting shall be achieved by simultaneously exposing the three cameras with their optical axes oriented to contain the C-camera image with the B-camera image and the B-camera image with the A-camera image. This nesting concept and image format is illustrated in Figure 1.

3.3.1.4 Operating Light Levels - Sensitivity

The cameras shall use standard vidicon tubes to operate over a range of surface illumination from 30 ft-Lamberts to 3000 ft-Lamberts and shall be designed to achieve unity signal-to-noise ratio at 30-ft Lamberts. Signal-to-nose ratio is defined in this context as the condition when vidicon signal is equal to the peak-to-peak noise. Vidicon γ must be between 0.65 and 0.70.

3.3.1.5 Spectral Response and Filters

The television images shall cover the visible spectrum from 4000 A to 7000 A with a peak spectral response between 4800 A and 5800A.

The use of 2-color filters on the low resolution A-camera shall be considered. If utilized, the filters shall be arranged as shown in Figure 1. Filter spectral response shall be chosen such that the combined spectral response of the vidicon and filters results in equal energy transmission on each portion of the filtered image. Peak response of the filter-vidicon combinations shall be between 4700 A - 5500 A and 5800 A - 6200 A respectively. Filter efficiency shall be greater than 0.25.

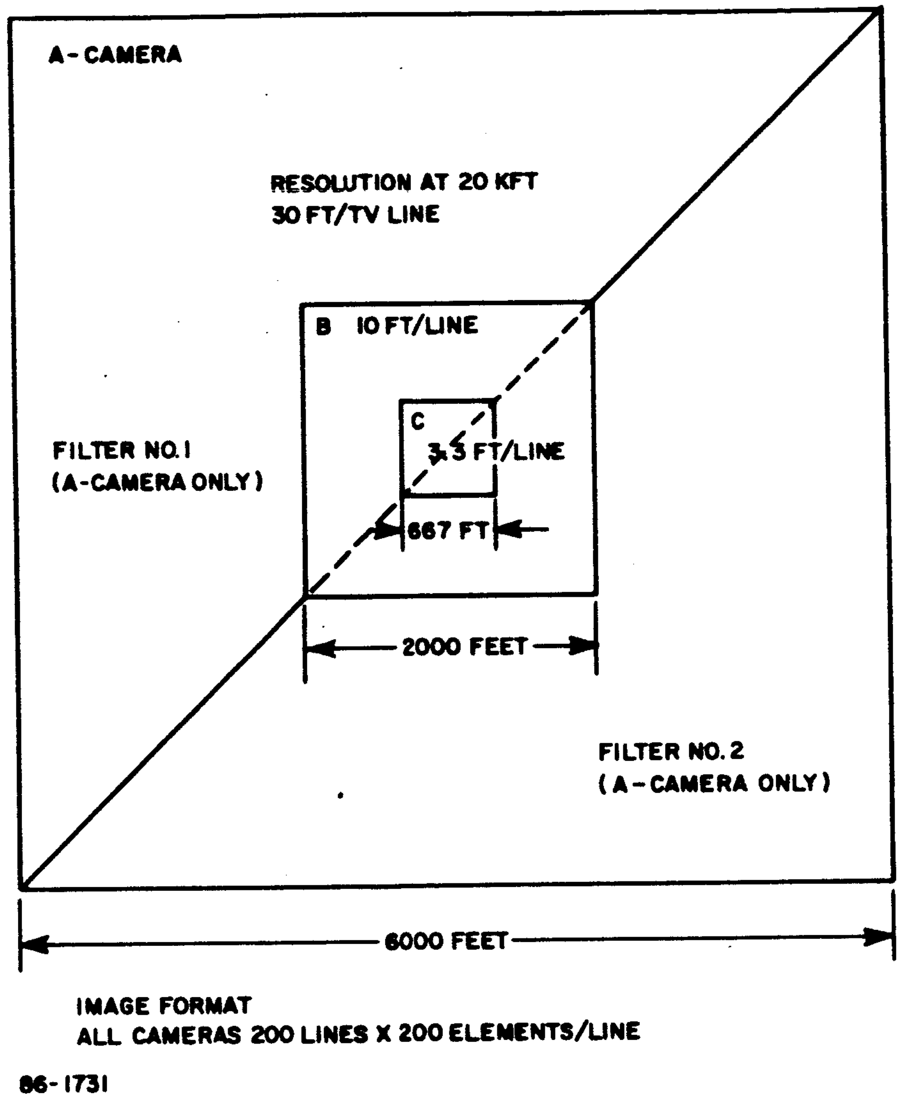


Figure 1 IMAGE FORMAT, RESOLUTION AND AREA COVERAGE FOR VERTICAL IMAGES AT 20,000 FEET

3.3.1.6 Camera Frame Duration

The cameras shall be designed for frame times of approximately 6 seconds and repetition intervals of 35 seconds. Suitable erasing mechanisms shall be provided to assure that the residual image after 35 seconds is less than 5 percent.

3.3.1.7 Shutters

Shutters shall be activated on external command. Double rotary blade mechanical focal plane shutters are required for each camera. Shutters shall be designed to cover an exposure range of 500 μsec to 5 msec. Shutters shall be set to nominal exposure durations to an accuracy of $\pm$ 1 percent and shall operate with $\pm$ 5 percent of the preset value during the mission duration. Each shutter shall incorporate a mechanical switch to confirm its correct operation. The three camera shutters shall be activated such that the shorter shutter openings occur during the longer shutter openings.

3.3.1.8 Electronics

The camera electronics shall provide for the independent generation of reading and primary scan signals, synchronization signals and blanking signals for each camera as well as suitable test points for diagnostic telemetry as indicated in paragraph 3.3.5.3.

3.3.1.9 Optics

Single lens fixed focal length optics shall be used with each camera. The optical systems for each camera shall be designed compatible with the vidicon tube to achieve the required sensitivity (paragraph 3.3.1.4) and resolution (paragraph 3.3.1.1). The combined efficiency of the optical system and protective window for each camera shall be equal to or better than 0.72 over the spectral response region of the vidicon.

3.3.2 Power Conditioning

The television subsystem shall provide power conditioning required to generate all supply and bias voltages for the functional elements of the TV subsystem. Primary power shall be supplied to the TV subsystem at 28 volts $\pm$ 1 v from a regulated Flight Capsule supply.

3.3.3 Camera Stabilization Platform

3.3.3.1 Requirements

The TV cameras shall be mounted on a stabilized platform to compensate for Flight Capsule motions in pitch and yaw. The stabilized platform is not required to compensate for image motion due to drift velocity over the surface, descent velocity or Flight Capsule roll. The camera stabilization platform is required to operate for Flight Capsule roll axis angles within 45 degrees of vertical. The Flight Capsule dynamic environment and camera stabilization requirements are given in Table I.

3.3.3.2 Relation to Inertial Reference Subsystem

The portion of the camera stabilization platform consisting of gimbals, torque motors, mounting fixtures, and gimbal release mechanisms shall be regarded as part of the TV subsystem. Drive signals shall be generated in the inertial reference subsystem.

3.3.4 Image Programming

The only external commands (in addition to platform control signals) to the TV subsystem shall be

1. Checkout/Calibrate (prior to or during flight)

2. Release Gimbals (after the parachute has been deployed)

3. Open Shutter (at appropriate intervals during the parachute descent phase of the Flight Capsule mission)

An image programmer shall be required in the television subsystem to interpret these commands into appropriate actuation and control sequences, and to perform internal housekeeping functions.

3.3.5 Data Generation and Processing

3.3.5.1 Video Data

The TV subsystem shall generate video data in the form of 40,000 - 5-bit words transmitted serially from each camera within 6 seconds of each shutter opening.

TABLE I

FLIGHT CAPSULE DYNAMICS AND CAMERA STABILIZATION REQUIREMENTS

Dynamics Variables	FC Dynamics	Camera Dynamics on Stabilized Platform
Altitude range	27, 500 to 1000 feet	27, 500 to 1000 feet
Horizontal drift Velocity (wind induced)	0 to 200 ft/sec	0 to 200 ft/sec
Horizontal drift Velocity (main parachute swing)	0 to 20 ft/sec	0 to 20 ft/sec
Vertical descent velocity	70 to 150 ft/sec	70 to 150 ft/sec
Roll rate	0 to 10 rpm	1 to 10 rpm
Pitch rate	0 to 100 degree/sec	<0.001 degree/sec for FC roll axis angle within 45 degrees of local vertical; $<$ FC pitch rate for FC roll angle $>$ 45 degrees
Yaw rate	0 to 100 degree/sec	<0.001 degree/sec for FC roll axis angle within 45 degrees at local vertical; $\leq$ FC yaw rate for FC roll angle $>$ 45 degrees
Roll axis angle	0 to 90 degrees from vertical	0 ± 1 degree from local vertical for FC roll axis angle <45 degree: FC roll axis angle -45 degrees for FC roll axis angle $>$ 45 degrees

The data shall be supplied on 5 parallel wires at a fixed rate of less than 75,000 words/second. Signal and impedance levels will be specified, but shall be less than 10.0 volts and 1 kilohms.

3.3.5.2 Photometric Data

Independent photometric measurements shall be made of the Martian surface by photocells mounted on the camera platform. Four types of measurements are required:

1. Wide band-wide angle measurements (field of view approximately 140 degrees, spectral response approximately equal to vidicon surface response)

2. Wide band-narrow angle measurements (field of view approximately equal to 25 degrees, spectral response as in 1.

3. Narrow band-narrow angle measurements (field of view approximately 25 degrees, spectral response approximately equal to filter No. 1-vidicon combination)

4. Narrow band-narrow angle measurements (field of view approximately 25 degrees, spectral response approximately equal to filter 2 - vidicon combination).

The photocell-filter-lens combination shall measure average surface brightness to ± 5 percent.

3.3.5.3 Diagnostic Data

The TV subsystem shall provide test points suitable for generating the following diagnostic and engineering telemetry:

Event Telemetry (A-refers to A-Camera, etc)

1. Shutter confirmation (A)

2. Shutter confirmation (B)

3. Shutter confirmation (C)

4. Gimbal release confirmation

5. Shutter enable gate set pulse

Signal Telemetry (A-refers to A-Camera, etc.)

1. Beam current (A)
2. Beam current (B)
3. Beam current (C)
4. AGC voltage (A)
5. AGC voltage (B)
6. AGC voltage (C)
7. Clamp (black) level (A)
8. Clamp (black) Level (B)
9. Clamp (black) level (C)
10. High voltage (A)
11. High voltage (B)
12. High voltage (C)
13. Target voltage (A)
14. Target voltage (B)
15. Target voltage (C)
16. Erase lamp current
17. TV package temperature
18. Heater current
19. TV package pressure
20. Photocell 1 current
21. Photocell 2 current
22. Photocell 3 current
23. Photocell 4 current

3.3.6 Environmental Packaging

The TV subsystem shall be packaged and housed to eliminate operating hazards due to:

1. High voltage breakdown
2. Debris generated from separation operations
3. Thermal transients
4. Low storage temperatures.

Active thermal control, protective windows and pressurization shall be used where required to achieve protection from these and other specific hazards.

3.3.7 Mechanical Characteristics

TV subsystem weight is limited to 60 pounds with a maximum weight of 35 pounds on platform gimbals.

3.3.8 Power Consumption

The TV subsystem shall dissipate less than 27 watts continuous dc power at 28 ± 1 volt during its operating interval from Flight Capsule separation to impact (less than 2 hours). At all other times the TV subsystem may dissipate up to 5 watts intermittently.

3.4 Interfaces

3.4.1 Separation Subsystem

The separation subsystem shall provide a viewing port of sufficient size to avoid covering the field of view of the television cameras at altitudes lower than 27,500 feet and Mach numbers lower than 1.2.

In the primary operating mode (with heat shield and entry shell separated) the camera viewing port shall cover at least 52 degrees solid angle centered around the vehicle roll axis and pointing downward. In the backup performance mode (heat shield and shell attached) a viewing port of at least 45 degrees shall be provided.

3.4.2 CC & S Subsystem

3.4.2.1 Inputs

The CC & S subsystem shall control the television experiment through the following commands:

1. Checkout (prior to or during flight)

2. Calibrate (prior to or during flight)

3. Release gimbals (after the parachute has been deployed)

4. Open Shutter (at appropriate times during the parachute descent phase).

   The release gimbals and open shutter commands shall be supplied by the CC & S.

   In addition, the CC & S system will provide a standard clock pulse at a convenient frequency between 1 and 100 kilocycles internal timing of the TV subsystem.

3.4.2.2 Outputs

The TV subsystem will provide two signals to the CC & S system:

1. Shutter Confirmation (when the shutters are fired).

2. Gimbal Release Confirmation (indicating that the gimbals are both mechanically free to respond to IRS commands.

3.4.3 Inertial Reference Subsystem

The IRS shall provide Functional Checkout attitude data and develop control signals required for camera stabilization as follows:

1. The IRS computer shall provide digital indications of the vehicle roll axis look angle with respect to the vertical θ_c and the Suspended Capsule swing rate, $\dot{\theta}_c$ as follows:

$|\theta_c| \leq 30 \pm 3$ degrees

$|\dot{\theta}_c| \leq 13 \pm 1$ deg/sec

2. The IRS shall provide gimbal control signals required for camera stabilization. A two-axis gimbal system is required to stabilize the camera in pitch and yaw. Roll stabilization is not required. The gimbal control signals shall result in maintaining the camera optical axis within 1 degree of local vertical, and reducing camera pitch and yaw rates to less than 0.001 deg/sec. The gimbal control signals shall result in this performance for Suspended Capsule roll axis angles within 45 degrees of local vertical.

3.4.4 Telecommunications

The telecommunication subsystem shall provide for the storage and transmission of digital video data and for the generation of video identification data other than line and frame data.

Analog-to-digital converters in the TV subsystem shall supply parallel 5-bit video signals to digital memories in the telecommunications system. Each camera shall generate approximately 40,000 5-bit words in 6 second read-out periods which occur at least 28.4 seconds apart. These data shall be generated in synchronism with a read sync signal by telecommunications. The telecommunications system must insure that pictures are transmitted in the order CBA CBA CBA...............

3.4.5 Diagnostic Data Acquisition

The diagnostic data acquisition area shall provide for sampling and conditioning certain analog and event data required to assess the operation of the television system.

Diagnostic telemetry requirements are listed in paragraph 3.3.5.3.

BATTERY COMPONENT SPECIFICATION

Number III-2-2-J

1.0 SCOPE

This document specifies the design and performance requirements of the battery as a component of the Flight Capsule electrical power and control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Power Control, Component Specification, Number III-2-2-K

2.3 Load Voltage Regulator, Component Specification, Number III-2-2-L

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Functional Description

The battery shall be a secondary nickel-cadmium unit consisting of not more than 24 hermetically sealed cells connected in series and held together in a suitable container.

3.2 Operation

3.2.1 Capacity

The battery shall provide a minimum of 740 watt-hours at a discharge temperature of +40° F when discharged at the rate shown in Figure 1.

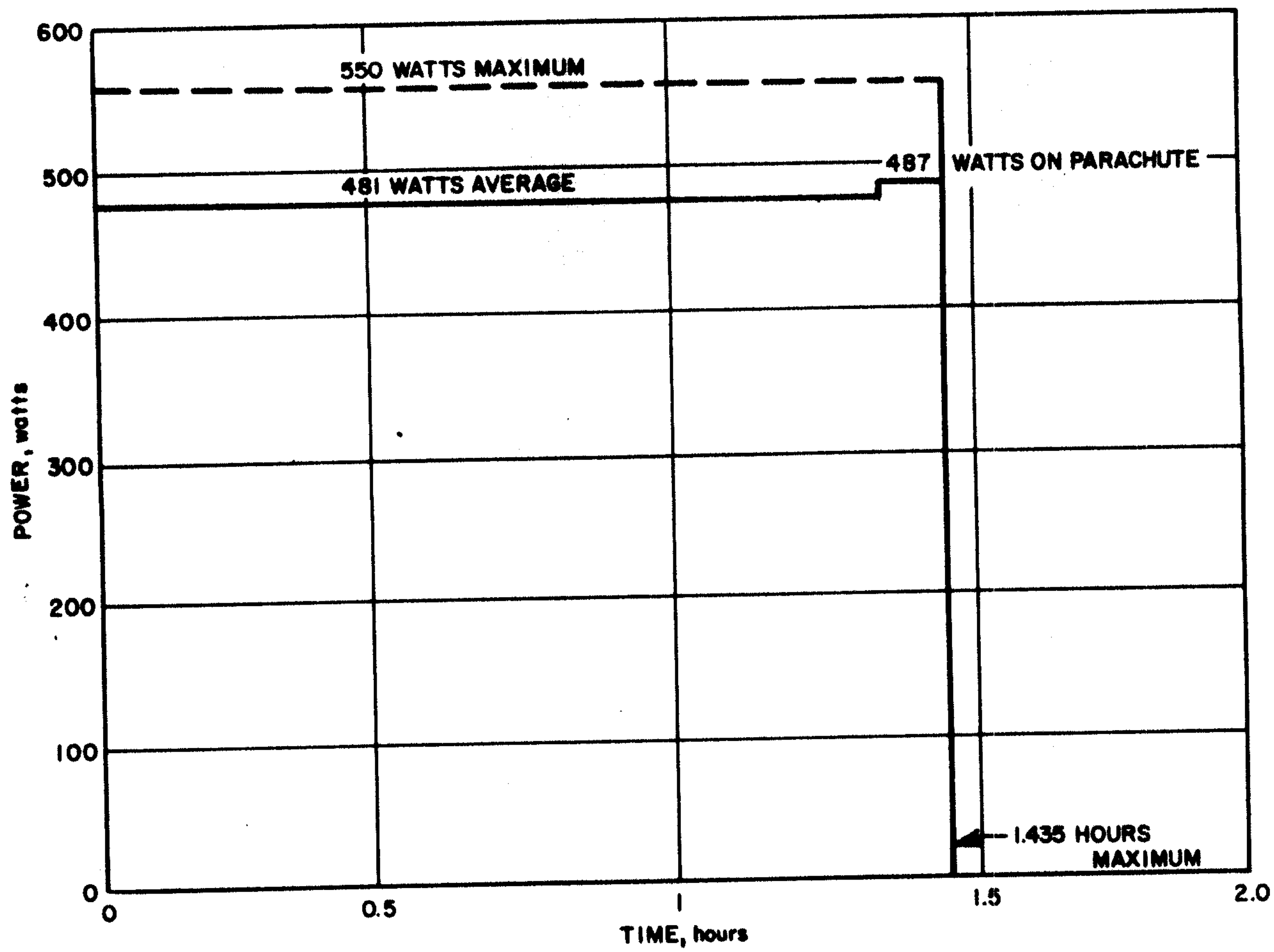
3.2.2 Output Voltages

Output voltages shall be as follows:

22 vdc to 32 vdc under load

28.5 vdc average plateau

35.0 vdc maximum open circuit



86-1732

Figure 1 POWER PROFILE FOR FLIGHT CAPSULE

3.2.3 Charge-Discharge Cycles

The battery shall be capable of performing a minimum of 10 full-depth discharge-charge cycles at the rates shown in Figure 1. At least 5 of these cycles must be performed after full heat sterilization specified in 2.1.

3.2.4 Charging

1. The battery shall be capable of accepting a continuous trickle charge at a rate of C/100 indefinitely at the operating temperatures in paragraph 3.2.6.

2. The battery shall be capable of being fully charged in a maximum period of 14 hours after a discharge according to Figure 1.

3. Minimum charge efficiency shall be 70 percent.

3.2.5 Operating Attitude

The battery shall be capable of being charged and discharged in any attitude.

3.2.6 Operating Temperatures

The battery shall be operative after exposure to the following temperature:

1. Launch to Separation: - +40 to +120°F

2. Separation thru Impact: - +40 to +200°F

3.3 Characteristics

3.3.1 Operating Environments: As specified in 2.1.

3.3.2 Nonoperating Temperatures:

Storage (charged): -100 to +300°F

3.3.3 Heat Sterilization:

1. The battery shall be heat sterilized with all cells in a fully discharged and shorted condition. A shorting circuits shall connect each positive and negative terminal. Provision shall be made for shorting out each cell separately at the battery electrical connector.

2. Each battery cell shall be capable of withstanding a minimum of 6 heat sterilization cycles for a 36 hour thermal soak at +145°C.

3.3.4 Weight and Size

The battery shall not exceed a weight of 53 pounds. The maximum volume occupied by the battery (including case) shall be 790 in^3 in a rectangular configuration.

3.3.5 Cell Construction

Each cell shall be hermetically sealed in a metal case utilizing a ceramic to metal or glass to metal seal capable of maintaining cell operation in a vacuum of 10^{-5} torr for a minimum of 1 year after final heat sterilization.

3.3.6 Magnetic Cleanliness

The design of each cell shall minimize the magnetic field generated by the battery.

3.4 Interfaces

3.4.1 Electrical Inputs

1. From The Battery Charger

   Current levels as specified in paragraph 3.2.4.

2. From The Launch Complex

   Current levels as specified in paragraph 3.2.4.

3.4.2 Electrical Outputs

1. To the subsystem via the power control switches.

2. To the power converters (voltage regulator) via the power control switches.

3. To the power control switches via the CC&S.

4. To the telemetering subsystem (analog signals representing the following diagnostic data: battery terminal voltage, battery current (charging & discharging), and battery temperature).

3.4.3 Mechanical Interfaces

The battery and interconnecting electrical cables shall be attached to the Suspended Capsule support structure.

POWER CONTROL COMPONENT SPECIFICATION

Number III-2-2-K

1.0 SCOPE

This document specifies the design and performance requirements of the power control as a component of the Flight Capsule electrical power and control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Load Voltage Regulator, Component Specification, Number III-2-2-L.

2.3 Battery, Component Specification, Number III-2-2-J.

2.4 Military Specifications

2.4.1 MIL-R-6106-Relay, Electric, Aerospace, General Specification for.

2.4.2 MIL-T-5757D-Relays, Electric (Excluding Thermal) for Electronic and Communication Type Equipment, General Specification for.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Functional Description

The power control unit shall consist of switches, blocking diodes, and fuses arranged to perform the switching required, plus additional implementation to protect all users and the batteries from the effects of failure of any one part. All parts shall be enclosed in an hermetically sealed container. The effective boundaries of operation and logic shall be as shown in Figure 1.

3.2 Operation

3.2.1 Power Rating - The power control unit shall have a nominal rated power handling capacity of 550 watts continuous with provision for intermittent spikes of up to 700 watts.

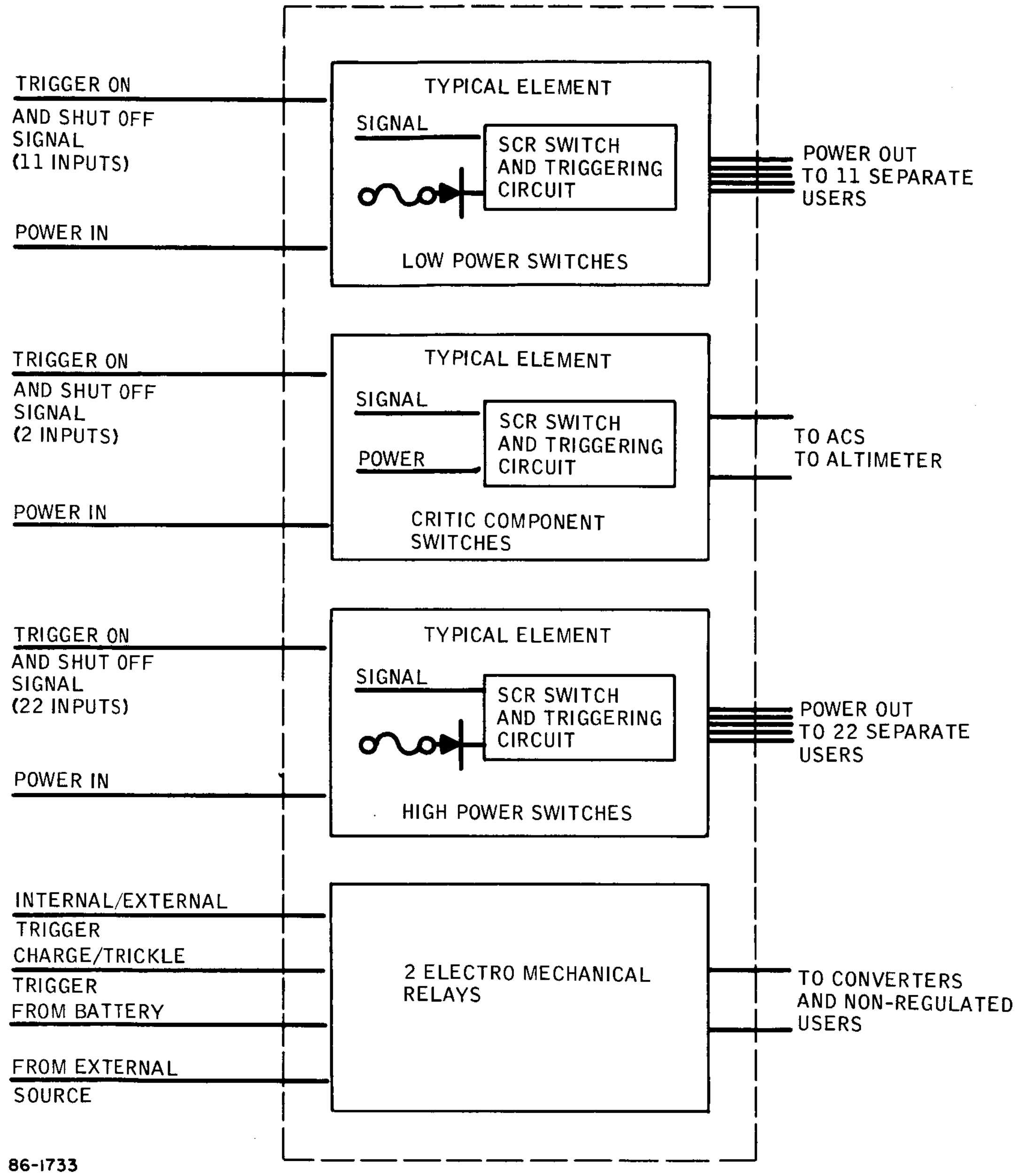


Figure 1 POWER CONTROL SUBSYSTEM

3.2.2 Turn On Times - Turn on times shall be as follows:

For low power users: <2 μsec with 3 volt signal.

For high power users: <3 μsec with 3 volt signal.

High power users are defined as switches having a current capacity of 5 amps or greater. Low power users are defined as having a maximum current capacity (continuous) of 1.1 amps.

3.2.3 Turn Off Times

Turn off times shall be a maximum of double the turn on times stated in paragraph 3.2.2.

3.2.4 Firing Signal

The nominal required firing (trigger) signal for any one switch shall not exceed 50 ma at 2 volts and + 20° C, 30 ma at 3 volts and + 20° C, and 20ma at 4 volts and + 20° C.

3.2.5 Maximum Forward Voltage Drops

Low Power: 1.1 volts at 1.1 amp

High Power: 0.8 volts at 5.5 amp

The total voltage drop in any one switching circuit shall be measured between the power control input and output terminals.

3.2.6 Number of Solid-State Switching Circuits

A minimum number of switching circuits shall be provided as follows:

Low Power: 13

High Power: 22

3.2.7 Blocking Diodes and Fuses

Blocking diodes and fuses shall be incorporated in all but two switching circuits. Blocking diodes shall be capable of blocking in excess of 400 volts. The fuses shall be of the slow-blow type and rated at 200 percent of the expected load in that circuit.

3.2.8 Switching Relays (Mechanical)

Two mechanical "latching type" relays shall be incorporated as shown in Figure 1. Each of these relays shall be capable of passing a constant power level of 550 watts with provision for intermittent spikes of up to 1 kilowatt. The relays shall be of the mechanical latching variety and capable of locking in both the energized or de-energized conditions as required until reset by an external electrical pulse to either coil. Both relays shall comply with 2.4.1. Pick-up voltages and coil resistance shall comply with 2.4.2. Pickup operating time shall not exceed 6 msec.

3.2.9 Thermal Dissipation

The maximum heat dissipation due to the power control unit shall be 50 watts.

3.2.10 Operating Attitude

Any

3.2.11 Operating Temperatures

Maximum sink temperature: +255°F

Minimum sink temperature: -85°F

Nonoperating temperatures: -85 to 300°F

3.3 General Characteristics

3.3.1 Operating Environments

As specified in 2.1.

3.3.2 Weight and Size

The maximum weight of the unit shall be 9 lbs. The maximum size shall be 200 in^3 and the shape shall be a rectangular box of which one dimension is approximately 6.25 inches.

3.3.3 Magnetic Cleanliness and RFI

The components shall be arranged in such a manner as to minimize RFI and magnetic field generation.

3.3.4 Externally Induced RFI

Provision shall be made in each switching circuit to prevent accidental triggering of solid state switches by short time RFI type pulse inputs.

3.4 Interfaces

3.4.1 Electrical inputs

All dc

1. Trigger and turn off signals for solid-state switches from the programming and sequencing unit (CC&S).

2. Trigger and turn off signals for mechanical switches and selected solid state switches from the launch complex and Flight Spacecraft.

3. Input power directly from the batteries (nonregulated).

4. Input power from converters (regulated).

5. Input power from external sources.

3.4.2 Electrical Outputs

All dc

1.	Low power:	13 low power lines (regulated)
2.	High power:	22 high power lines
3.	Main power:	2 lines from mechanical relays
4.	To CC&S:	power for operation

3.4.3 Mechanical Interfaces

The power control unit and interconnecting electrical cables shall be attached to the Suspended Capsule support structures.

LOAD VOLTAGE REGULATOR COMPONENT SPECIFICATION

Number III-2-2-L

1.0 SCOPE

This document specifies the design and performance requirements of the load voltage regulator as a component of the Flight Capsule electrical power and control subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Battery, Component Specification, Number III, 2-2-J.

2.3 Power Control, Component Specification, Number III-2-2-K.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Functional Description

The load voltage regulator unit shall accept dc power within a broad range of voltage and provide a closely voltage regulated dc power output. It shall consist of as many switching, amplifying and converter units as are necessary to perform the required function. The complete unit shall be contained in a hermetically sealed container equipped with suitable connectors and mounting flanges.

3.2 Operation

3.2.1 Power Rating

The regulator shall have a nominal continual power handling capability of 550 watts with provision for intermittent power of up to 700 watts.

3.2.2 Voltage and Current Loading

Input:	22 to 35 vdc
Output:	28 vdc ± 1 percent
Current Output:	0.5 to 20 amps continuous and up to 25 amps intermittent

Instantaneous
Current Overload: 100 percent

3.2.3 Output Characteristics

Ripple:	Less than 0.10 volts peak-to-peak
Transients:	Less than 0.75 volts for 7 amp load changes
Switch Voltage Sensitivity and Stability:	± 0.050 volts from -40 to +170°F

3.2.4 Operating Characteristics -- Buck-Boost

3.2.5 Operating Efficiency

Buck Mode:	85 to 95 percent
Boost Mode:	85 to 90 percent
Direct Mode:	98 to 100 percent

3.2.6 Operating Attitude -- any

3.2.7 Operating Temperatures -- 40 to +160°F and to 200°F transients.

3.3 General Characteristics

3.3.1 Environment

The voltage regulator shall be operable after exposure to the operating and nonoperating environments of 2.1.

3.3.2 Temperature

The voltage regulator shall be operable after exposure to a nonoperating temperature of -85 to +300°F.

3.3.3 Heat Sterilization

The voltage regulator shall be capable of performing its functions after subjection to the heat sterilization requirements listed in 2.1.

3.3.4 Weight and Size

1. The maximum weight shall be 12 pounds

2. The maximum size shall be 300 in^3 of which one dimension is approximately 6.25 inches.

3.3.5 Magnetic Cleanliness and RFI

The components shall be arranged and shielded in such a manner so as to minimize RFI (both externally induced and internally generated) and minimize magnetic field generation in all directions.

3.4 Interfaces

3.4.1 Electrical Inputs

1. From the batteries

2. From external power

3.4.2 Electrical Outputs

To all subsystems as required.

3.4.3 Mechanical Interfaces

The voltage regulator and the interconnecting electrical cables shall be attached to the Suspended Capsule support structure.

3.4.4 Thermal

A maximum of 125 watts shall be dissipated continuously for 5 hours from preseparation checkout to impact.

RADIO SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-M

1.0 SCOPE

This document specifies the performance and design requirements of the radio subsystem as a component of the Flight Capsule tele-communications subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Data Handling and Storage Subsystem, Component Specification, Number III-2-2-N.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

3.1.1 General

The basic components of the radio subsystem include the following:

1. Antenna

2. Transmitter

3.1.2 Function

The Flight Capsule radio subsystem shall perform the following functions:

1. Receive the serial binary coded, nonreturn to zero (NRZ) telemetry signal from the data handling and storage subsystem.

2. Modulate the transmitted RF signal with the telemetry signal.

3. Transmit the modulated RF signal to the Flight Spacecraft.

3.2 Operation

The radio subsystem shall be turned on periodically during status checks of the Flight Capsule subsystem prior to separation from the Flight Spacecraft. Shortly before separation, the radio subsystem shall be turned on and shall remain operational for the completion of the mission.

3.3 Characteristics

3.3.1 Frequency

The carrier frequency shall be in the 267 to 273 megacycles band.

3.3.2 Radiated Power

The radiated power shall be 30 watts minimum. During entry 30 watts shall be reflected back into the subsystem if radio blackout is experienced. After blackout the radio subsystem performance shall not be degraded from its level prior to blackout.

3.3.3 Modulation

The modulation technique shall be wideband noncoherent frequency shift keyed (FSK).

3.3.4 Stability

The frequency stability shall be ±0.005 percent maximum.

3.3.5 Antenna Polarization

The antenna shall be circularly polarized.

3.3.6 Antenna Gain

The antenna gain shall be -14 db minimum in a 270 degree beamwidth centered along the Entry Vehicle roll axis.

3.3.7 Antenna Beamwidth

The antenna beamwidth shall be 270 degree minimum.

3.3.8 Corona and Arcing

Circuits involving potentials in excess of 50 volts shall be suitably sealed to prevent arc over or corona discharge in a rarefied atmosphere.

3.4 Interfaces

3.4.1 Electrical

3.4.1.1 Power

The radio subsystem shall be capable of operating from a nominal 28 vdc source at a power consumption of 170 watts maximum.

3.4.1.2 Signal

The data handling subsystem shall provide the serial binary coded NRZ telemetry waveform for modulating the transmitter.

3.4.2 Mechanical

3.4.2.1 Weight

The weight of the radio subsystem shall not exceed 29 pounds.

3.4.2.2 Volume

The volume of the radio subsystem shall not exceed 6000 in^3.

3.4.2.3 Envelope Restrictions

The antenna shape shall be cylindrical, 18 inches in diameter by 11 inches deep.

3.4.4 Thermal

The radio subsystem will be maintained within temperature extremes of -20 to +80°C operating and -55 to +135°C non-operating.

DATA HANDLING AND STORAGE SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-N

1.0 SCOPE

This document specifies the performance and the design requirements of the data handling and storage subsystem as a component of the Flight Capsule telecommunications subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

2.2 Radio Subsystem, Component Specification, Number III-2-2-M

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

3.1.1 General

The basic elements of the data handling and storage subsystem (DH&SS) shall include the following:

1. Engineering Data Handling Subsystem
2. Diagnostic Data Handling Subsystem
3. Data Storage Subsystem.

3.1.2 Function

The data handling and storage subsystem shall perform the following functions:

1. Control and synchronize the sampling of the instruments to establish the instrument internal sequence.
2. Perform the necessary conversions and encoding of the several forms of engineering and diagnostic data.
3. Buffer the engineering and diagnostic data which occurs at different and sporadic rates and make them available at various but constant desired rates to the time multiplexer.
4. Use the binary coded and time multiplexed engineering and diagnostic data to frequency shift key (FSK) modulate the RF carrier of the Flight Capsule to Flight Spacecraft telemetry link.

5. Provide capability for storage of the TV data.
6. Provide a time delayed (stored) replica of certain diagnostic and experiment data (binary form) for inclusion in the time multiplexed sequence.

3.2 Operation

3.2.1 Data Acquisition

The data handling subsystem shall control engineering and diagnostic data collected for all the instruments in the Flight Capsule. The list of measurements and operational times during which data collection is required are shown in Table I and Figure 1, respectively.

All measurements in the general experiment category shall be sampled simultaneously in order to provide time correlation between the measurements.

All diagnostic and engineering data that is acquired during entry shall also be stored in a delay memory and played out again after end of blackout (EBO) and prior to impact. This technique insures that any data loss during entry due to Entry Vehicle coning or radio blackout will be retransmitted after emergence from blackout.

3.2.2 Data Playout

The TV playout data rate shall satisfy the requirement to completely transmit three sets of TV pictures (nine pictures) in a minimum time of 150 seconds.

The penetrometer data samples shall be characterized by a 10-bit word; 6-bit accuracy and 4-bit identification. The playout rate requirement for the penetrometer data is 50 samples from each output in a 7.5-second interval. This requirement guarantees the return of data from the first two shock pulses from each penetrometer before the Suspended Capsule impacts.

3.3 Summary of Characteristics

3.3.1 Channel Capacity

The minimum number of data handling subsystem channel inputs shall be:

Analog - 384
Digital - 79

TABLE I

DATA REQUIREMENTS LIST

Measurement	Sampling Rate	Accuracy	Type of Output	No. of Output
Engineering				
TV	7.5 kc/s(1)	5 bits	digital	3
Penetrometer	10 kc/s(2)	0.1 msec	analog	4
Radar altimeter Surface roughness	150 sps	7 bits	digital	1
General Experiments (3,4)				
Calibration	1 sps	0.1%	analog	3
Accelerometers	1 sps	0.1%	analog	6
Temperature	1 sps	1%	analog	2
Pressure	1 sps	1%	analog	2
Water detector	1 sps	1%	analog	1
Velocity attitude sensor	1 sps	1%	analog	10
Beta Scatter	1 sps	14 bits	count	1
Radar altimeter range	1 sps	7 bits	digital	1
Radar altimeter range rate	1 sps	7 bits	digital	1
Gas chromatograph	0.2 sps	1%	analog	8
Acoustic	0.2 sps	1%	analog	3
Mass spectrometer	0.2 sps	1%	analog	160
Radar detector	0.2 sps	1%	count	4
Diagnostic(3)				
No. 1 - 109	0.2 sps	5%	analog	109
No. 110 - 113	0.2 sps	2%	analog	4
No. 114 - 181	0.2 sps	1%	analog	68
No. 182 - 188	0.2 sps	0.1%	analog	7
No. 189 - 200	0.2 sps	± 2 sec	digital	12
No. 201 - 245	0.2 sps	± 2 sec	digital	45
No. 246 - 256	0.2 sps	± 0.5 sec	digital	11

1. Each set of three pictures recorded on the image tubes in the TV subsystem shall be erased simultaneously in six seconds. The resulting data (200 x 200 elements 5 bit digital words per picture) shall be incorporated into the transmitted data such that a minimum of three sets (9 pictures) may be taken at equal time intervals during descent.
2. Maximum penetrometer sampling period is 60 msec.
3. This data must also be delayed and retransmitted 100 seconds later.
4. All instruments in this category are sampled simultaneously.

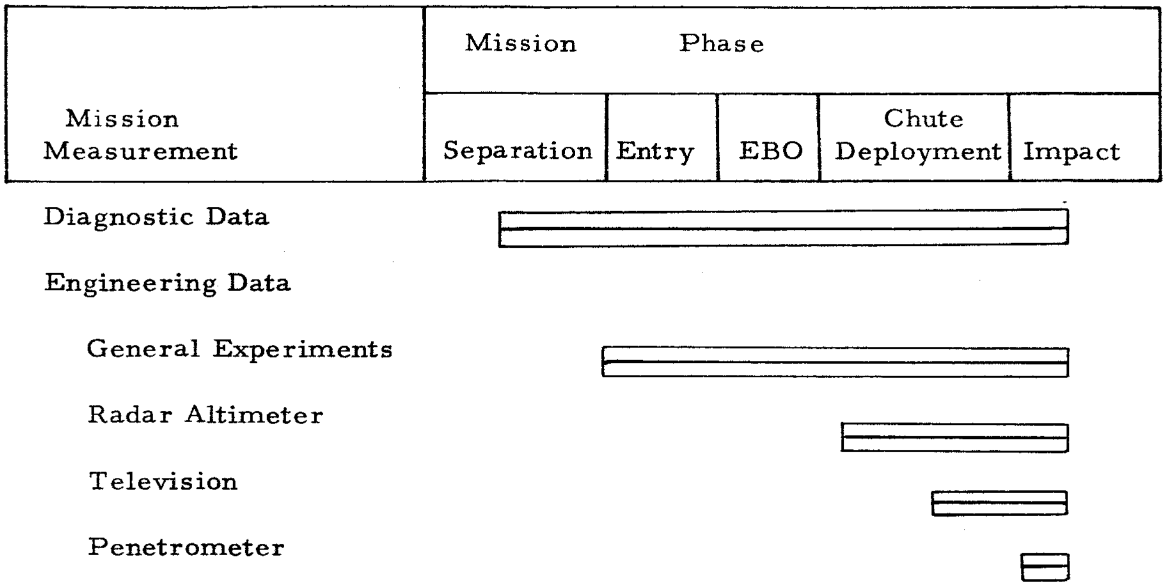


Figure 1 REQUIRED DATA COLLECTION PROFILE

3.3.2 Data Rate

The rate of pulse code modulated (PCM) nonreturn to zero (NRZ) data to the radio subsystem shall be 18,000 bps maximum.

3.3.3 Accuracy

The measurement accuracy shall be as indicated in Table 1.

3.3.4 Sampling Rates

The sampling rate of the measurements shall be as indicated in Table 1.

3.3.5 Input Signal Bandwidth

In order to satisfy the requirement for essentially simultaneous sampling of all the instruments in the general experiment category, all the data in any one sampling period shall be collected within 50 msec maximum.

3.3.6 Frame Length

The frame length shall be 1064 bits.

3.3.7 Word Length

The data word length shall be 7 bits nominal.

3.3.8 Storage Capacity

1. The TV storage capacity shall be sufficient to store 4 TV pictures at any one time.
2. The delay (blackout) storage capacity shall be sufficient to store all diagnostic and general experiment data for 100 seconds.

3.3.9 Synchronization Pattern

A 31-bit pseudo noise (PN) code shall serve as frame synchronization and shall precede the data in each frame.

3.3.10 Frame Identification

A 4-bit sync word shall follow the 31-bit synchronization pattern and shall serve to identify the data appearing in each frame.

3.3.11 Time Diversity

All data shall be played out twice at least 2.5 seconds apart to provide time diversity in the data.

3.3.12 Frame Format

The frame format shall provide for maximum allowance for failures and the minimum loss of data.

3.4 Interfaces

3.4.1 Signal

3.4.1.1 Electric Power and Control

The electric power and control subsystem shall provide nominal 28 vdc as required by the data handling and storage subsystem.

3.4.1.2 Command Control

The central computer and sequencer shall provide timing signals to control the operation of the data handling and storage subsystem.

3.4.1.3 Experiment Instrumentation

The engineering instrumentation shall provide the following inputs to the DH&SS:

1.	Analog inputs:	0-5 volts
2.	Digital event data:	1-bit word
3.	Digital data:	7- and 14-bit words

3.4.1.4 Radio Subsystem

The DH&SS shall provide a constant rate of PCM NRZ data to the radio subsystem.

3.4.2 Weight

The weight of the DH&SS shall not exceed 20 pounds.

3.4.3 Volume

The volume of the DH&SS shall not exceed 700 in^3.

3.4.4 Power

The power requirements of the DD&HS shall not exceed 10 watts.

3.4.5 Thermal

The data handling and storage subsystem shall be maintained within temperature extremes of -20 and +80°C operating and -55 and +135°C nonoperating.

PARACHUTE SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-P

1.0 SCOPE

This document specifies the performance and design requirements of the parachute as a subsystem of the Flight Capsule.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Description

The parachute subsystem shall consist of a ring-slot pilot parachute, a ring-sail main parachute, a pilot parachute canister and mortor assembly, a main parachute canister and gas generator assembly, and a swivel and harness attachment assembly. The parachute canopies shall be constructed of type 330 heat resistent nylon. The suspension lines shall be nylon cord and the riser line shall be nylon webbing. The overall parachute design shall be capable of withstanding a maximum opening dynamic pressure of 5.0 lb/ft^2 and a deployment Mach number of 1.2.

3.2 Operation

The central computer and sequencer (CC & S) shall provide a parachute deployment signal when the Entry Vehicle altitude is below 27,500 feet and the Mach number is less than 1.2. Both of these conditions shall be satisfied for initiation to take place. On receipt of the signal from CC & S, an electrical squib shall ignite a mortar charge which forces a sabot to eject the pilot parachute from its canister at 100 ft/sec. On inflation the pilot parachute shall pull the main parachute out of its canister in the deployment bag. At line stretch, the deployment bag shall be stripped off and full inflation of the main parachute shall occur.

In the event that the mortar does not eject the pilot parachute, a gas generator shall be ignited which in turn shall inflate a bag to force the main parachute out of its canister at 30 ft/sec. A swivel, which is located between the main parachute riser line and the harness assembly confluence point, shall be utilized to reduce the possibility of vehicle spin being induced into the main parachute shroud lines. A tension switch shall be located in the riser line to indicate main parachute inflation.

3.3 Characteristics

3.3.1 Component Weight

The weight breakdown of the subsystem and components shall not exceed the values tabulated below:

Component	Unit weights (pounds)	No. req'd	Weights (pounds)
Main parachute canopy	28.1	1	28.1
Main parachute suspension lines	0.6	48	28.8
Main parachute riser line and collars	2.6	1	2.6
Pilot parachute (canopy & lines)	1.0	1	1.0
Main parachute canister assembly (gas generator bag and explosive charge)	4.5	1	4.5
Pilot parachute canister assembly (sabot and explosive charge)	1.9	1	1.9
Tension switch	0.5	1	0.5
Swivel	2.0	1	2.0
Harness attachment assembly	2.3	1	2.3
		Total Subsystem Weight	71.7 lbs

3.3.2 Component Design

The subsystem components shall conform to the following:

1. Main Parachute Canopy - 81-foot nominal diameter ring-sail configuration constructed of nylon material

2. Main Parachute Suspension Lines - 93 feet in length capable of withstanding approximately 500 pounds of load each

3. Main Parachute Riser Line - 8 nylon web risers of 10 foot length

4. Pilot Parachute - 9-foot nominal diameter ring-slot configuration constructed of nylon material

5. Main Parachute Canister - Cylindrical shell of approximately 2.0 cubic feet internal volume - shell to be constructed of 9/32-inch gage aluminum or magnesium.

6. Gas Generator Bag and Charge - Bag volume: approximately 2.0 cubic feet - Bag material: nomex or equivalent

7. Pilot Parachute Canister - Cylindrical shell 3 inches in diameter by 6 inches in length to be constructed of 9/32-inch gage aluminum or magnesium

8. Pilot Parachute Mortar Assembly - Standard sabot and mortar breech assembly capable of ejecting the pilot parachute at 100 ft/sec.

9. Tension Switch - To be defined

10. Swivel - Low friction swivel capable of withstanding opening shock load of main parachute (32,200 pounds including 1.7 factor of safety) - the overall length is approximately 8 inches.

11. Harness Attachment Assembly - Four nylong lines approximately 8 foot long (each) connected from the swivel to the Suspended Payload attachment points - tensile strength: 16,100 pounds each.

3.4 Interfaces

3.4.1 Electrical

1. Tension Switch Output - Switch closure when approximately 20,000 pounds load is applied to the swivel.

2. Continuity Loop Output for Pyrotechnics - Two for mortar initiation and two for the gas generator.

3. 4.5 amp signal input to eject pilot parachute.

4. 4.5 amp backup signal input to deploy main parachute.

3.4.2 Mechanical

1. Main parachute canister structural mounting

2. Main parachute harness attachment to the Suspended Capsule.

3.4.3 Thermal

Passive means to maintain the temperature within the desired range during space environment.

RADAR SUBSYSTEM COMPONENT SPECIFICATION

Number III-2-2-R

1.0 SCOPE

This document specifies the performance and design requirements of the radar subsystem as a component of the Flight Capsule programming and sequencing subsystem as well as the Flight Capsule scientific data acquisition subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Subsystem Description

The basic elements of the radar subsystem shall be:

1. Radar altimeter including both high altitude and low altitude altimeters including high and low altitude altimeter antennas.

2. Three beam velocity - altitude sensor

The radar subsystem shall perform the following functions:

1. Measure altitude from 250,000 feet down to approximately 10 feet

2. Determine the horizontal velocity of the Suspended Capsule with respect to the planet surface.

3. Obtain an estimate of the planet surface roughness.

4. Determine the descent attitude of the Suspended Capsule.

5. Provide altitude data to the central computer and sequencer (CC&S) to be used to initiate deployment of the penetrometers.

6. Serve as backup for initiation of entry shell jettisoning and parachute deployment

3.2 Operation

3.2.1 High Altitude Altimeter

1. The high altitude altimeter shall measure altitude from from 250,000 to 25,000 feet. The altimeter subsystem shall also supply measurements from which surface roughness can be estimated.

2. The high altitude altimeter shall operate at a frequency of 18 mc and utilize a transmitter power of not less than 25 watts. The antenna system shall provide a coverage of at least ± 75 degrees from the Flight Capsule roll axis.

3. The transmitter shall be modulated with an intermittent continuous wave (ICW) whose pulse repetition frequency is controlled by the altitude and varies from kilocycle at 250,000 feet to 83 kilocycles at 25,000 feet.

4. The altimeter subsystem shall provide a signal which represents the ratio of power in the diffuse echo to power in the specular return. These data provide an indication of furface roughness. A minimum of two such measurements shall be made during the high altitude phase.

3.2.2 Low Altitude Altimeter

1. The low altitude altimeter shall measure height from an altitude of 30,000 feet to an altitude of 1000 feet. Surface roughness measurements and wind determination shall also be provided.

2. The low altitude altimeter shall share common video and data processing circuitry with the high altitude altimeter.

3. The low altitude altimeter shall be used as the primary means for deploying the penetrometer.

4. The low altitude altimeter shall operate at a frequency of 324 mc and utilize a transmitter power of not less than one watt.

5. The antenna system must provide a coverage of ± 90 degrees with respect to the suspended payload vertical axis. The antenna shall also be capable of receiving signals from four penetrometers operating near 430 mc.

6. A diplexer shall separate the altimeter signals from the penetrometer signals.

7. The transmitter shall be modulated with ICW whose pulse repetition frequency is controlled by the altitude and varies from 93 kilocycles at 30,000 feet to 250 kilocycles at 1000 feet.

8. A surface roughness measurement shall be made at least twice during the low altitude phase similar to that performed by the high altitude altimeter.

9. The Doppler frequency spectrum shall provide an indication of wind velocity.

3.2.3 Velocity - Attitude Sensor

1. The velocity-attitude (VA) sensor shall function from an altitude of 30,000 feet to 10 feet. It shall determine the vertical and horizontal components of the Suspended Capsule motion with respect to the planet surface.

2. The VA sensor shall also determine the attitude of Suspended Payload for correction of the wind measurement.

3.3 Characteristics

3.3.1 High Altitude Altimeter

The following is a summary of the principle characteristics of the high altitude altimeter:

3.3.1.1 Transmitter Power

The transmitter power shall be 25 watts.

3.3.1.2 Operating Frequency

The operating frequency shall be 18 mc.

3.3.1.3 Altitude Regime

The altitude regime shall be from 250,000 to 25,000 feet.

3.3.1.4 Accuracy

The altitude measurement accuracy shall be within 5 percent.

3.3.1.5 Calibration

The high attitude altimeter shall contain provision for self-calibration.

3.3.1.6 Description of Outputs

Measurement	Sampling Run	No. Samples	Work Length
Altitude	1 sps	Continuous	9 bit
Surface roughness	150 sps	at least two	7 bit

3.3.1.7 Antenna

The antenna shall provide hemispheric coverage of not less than -6db at ± 70 degrees from the Suspended Capsule roll axis. Polarization shall be linear. The Flight Capsule entry shell shall be excited as the antenna.

3.3.2 Low-Altitude Altimeter

The following is a summary of the principle characteristics of the low altitude altimeter.

3.3.3.1 Radiated Power

The radiated power shall be not less than one watt.

3.3.2.2 Operating Frequency

The operating frequency shall be 324 mc.

3.3.2.3 Altitude Regime

The altitude regime shall be from 30,000 to 1,000 feet.

3.3.2.4 Accuracy

The altitude measurement accuracy shall be within 5 percent.

3.3.2.5 Description of Outputs

Measurement	Sampling Rate	No. Samples	Word Length
Altitude	1 sps	continuous	9 bit
Surface roughness	150 sps	at least two	7 bit
Wind measurement	1 sps	at least five	7 bit

3.3.2.6 Antenna

The antenna shall consist of a planar logarithmic spiral antenna 15 inches diameter by 8 inches deep. Polarization shall be linear. The gain at 90 degrees from the Suspended Capsule roll axis shall not be less than -9db.

3.3.3 Velocity-Altitude Sensor

The following is a summary of the principle characteristics of the velocity-altitude sensor.

3.3.3.1 Radar Beams

The velocity-altitude sensor shall consist of a 3 beam radar that measures range and range rate along each beam. The beamwidth of each beam shall not exceed 5 degrees. One antenna shall produce two beams: one beam shall operate at 13,000 kmc and the other beam shall operate at at 13, 300 kmc. A separate antenna shall produce the third beam at 13 kmc.

3.3.3.2 Transmitter Modulation

The transmitters shall be modulated with FM-CW whose pulse repetition frequency is controlled by the altitude and varies from 5 cps at 30,000 feet to 1 kilocycles at 10 feet.

3.3.3.3 Wind Velocity

Wind Velocity shall be estimated by means of a measurement of Suspended Capsule velocity along each of the three beams related to the sway angle of the Suspended Capsule.

3.3.3.4 Radiated Power

The radiated power shall be 1 watt per beam minimum.

3.3.3.5 Altitude Regime

The altitude regime shall be from 30,000 to 10 feet.

3.3.3.6 Accuracy

The wind measurement accuracy shall be within ±25 percent at a Suspended Capsule sway angle of ±45 degrees. The attitude determination shall be ±5 percent at a Suspended Capsule sway angle of ± 45 degrees.

3.3.3.7 Description of Outputs

Measurement	Sampling Rate	No. Samples	Word Length
Range (3)	1 sps	Continuous	7 bits
Range rate (3)	1 sps	Continuous	7 bits
Signal strength (3)	1 sps	Continuous	7 bits
Cone angle (1)	1 sps	Continuous	7 bits

3.3.3.8 Antenna

The antenna shall consist of two separate structures. The first of these shall be a torroid 20 inches in length 13 inches wide and 8.5 inches in height. This structure shall contain a feed for both 13,000 mc and 13,300 mc.

Polarization shall be linear.

The second structure shall be a torroid 16 inches long 13 inches wide and 8.5 inches in height. This structure shall contain a single feed and operate at 13,000 mc. Polarization shall be linear.

The weight of the dual antenna shall not exceed 7 pounds, the second antenna weight shall not exceed 5 pounds.

3.4 Interface

3.4.1 High Altitude Altimeter

3.4.1.1 Electrical

1. Input from the electrical power and control subsystem: 60 watts at 28 vdc nominal.

2. Output to the central computer and sequencer: altitude and velocity signals.

3. Output to the engineering data handling subsystem: altitude and surface roughness signals.

3.4.1.2 Mechanical - Electronic Assembly

1. Weight: 8.5 pounds maximum

2. Volume: 250 in^3 maximum

3.4.2 Low Altitude Altimeter

3.4.2.1 Electrical

1. Input from the electrical power and control subsystem: 4 watts at 28 vdc moninal.

2. Output to the central computer and sequencer: altitude and velocity signals.

3. Output to the engineering and data handling subsystem: altitude, surface roughness and wind velocity signals.

3.4.2.2 Mechanical - Electronic Assemblies

	Weight (Max) (pounds)	Volume (Max) (in^3)
Electronics	2.5	50
Diplexer	1.5	2
Antenna	7.0	1410

3.4.3 Velocity - Altitude Sensor

3.4.3.1 Electrical

1. Input from the electrical power and control subsystem: 10 watts at 28 vdc during warmup and standby, 50 watts at 28 vdc nominal during operation.

2. Output to the programming and sequencing subsystem: ten analog (0 to 5 v) signals representing velocity (3), range (3), signal strength (3) and angle (1).

3.4.3.2 Mechanical

1. Weight: 21 pounds maximum

2. Volume: 3220 $in.^3$ maximum

The high altitude and low altitude radar electronics and calibration network shall be packaged in a single assembly.

ABLATIVE HEAT SHIELD COMPONENT SPECIFICATION

Number III-2-2-S

1.0 SCOPE

This document specifies the performance and design requirements of the ablative heat shield as a component of the Flight Capsule entry shell subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

2.2 Standards, Federal Test Methods, Standard 406, Plastics, Organic, General Specification, Test Methods.

2.3 American Society for Testing Materials Specifications

2.3.1 ASTM C177-45 Thermal Conductivity of Materials by Means of the Guarded Hot Plate

2.3.2 ASTM C18-52 Chemical Analysis of Refractions

2.4 Avco Specification, RAD-P-(to be assigned), Ablative Heat Shield, Flight Capsule Entry Shell, Application of,

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

3.1 Performance Criteria

The ablative heat shield, after each application in accordance with 2.4 shall be uniform quality and shall have the following characteristics:

3.1.1 Hardness

The hardness of the finished heat shield at 73.5 ± 2°F (23 ± 1°C) shall be initially _*_ Shore A minimum and after 15 seconds of continuously applied load shall be _*_ Shore A minimum.

3.1.2 Specific Gravity

The specific gravity of the finished heat shield material shall be from _*_ to _*_ at a temperature of 73.5 ± 2°F (23 ± 1°C).

3.1.3 Filler

The percent of ash, representing the amount of filler in the finished heat shield material shall be _*_ plus or minus _*_ percent. The composition of the ash shall be _*_ .

3.1.4 Tensile Strength

The tensile strength of the heat shield material, measured at _*_ ± 2° F, using a strain rate of _*_ in./in./min, shall be _*_ lb/in^2 minimum.

3.1.5 Aging Characteristics

The aging characteristics of the heat shield material, prepared in accordance with 2.4 and stored for a period of 2 years, shall meet the requirement of 3.1.1, 3.1.2, 3.1.3 and 3.1.4.

3.2 Design Criteria

The heat shield material shall be tested for the following preproduction requirements. A recheck of these properties shall be required if the application process is altered in any way so as not to conform to 2.4.

3.2.1 Tensile Strength

The minimum tensile strength of the heat shield material measured at 300, 75 and -100° F using a strain rate of 0.05 in./in./min shall be 290, 330 and 868 lb/in^2, respectively.

3.2.2 Elastic Modulus

The secant modulus of elasticity at -100° F and a strain of 1.2 percent shall be 28 x 10^4 lb/in^2 minimum. The strain to failure at minus 100° F shall exceed 5.4 percent. The thermal strain from +300 to -100° F shall be a minimum of 0.018 in./in.

3.2.3 Thermal Conductivity

The material thermal conductivity at 250° F shall be 0.049 plus or minus 10 percent Btu ft/hr/ft^2/° F.

3.2.4 Specific Heat

The material Specific heat at 250° F shall be 0.34 plus or minus 10 percent Btu/lb.

3.2.5 Surface Temperature

The material surface temperature shall not exceed 1400°F.

3.2.6 Ablative Characteristics

The ratio of cold wall heat flux to the product of the mass loss and gas enthalpy shall be approximately equal to 1.45 for test conditions of enthalpy H_o = 6500 Btu/lb and heat flux q_c = 1400 Btu/ft^2/sec.

3.3 Operation

The heat shield material shall protect the entry shell and appropriate areas of the afterbody from atmospheric entry heating during the Entry Vehicle descent.

3.4 Interfaces

The heat shield material shall be attached to the entry shell and appropriate areas of the afterbody in various thicknesses dependent on the level of thermal protection required.

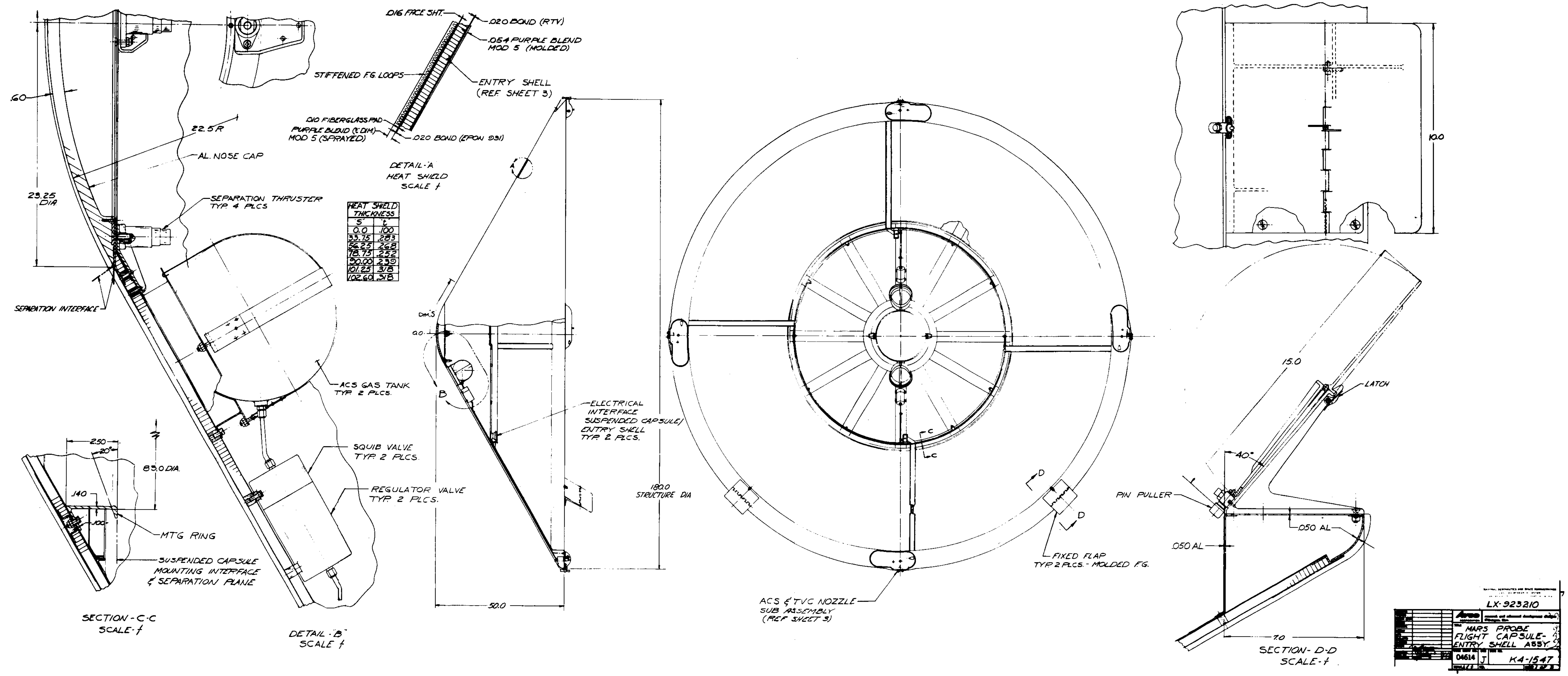


Figure 1 MARS PROBE FLIGHT CAPSULE -- ENTRY SHELL ASSEMBLY

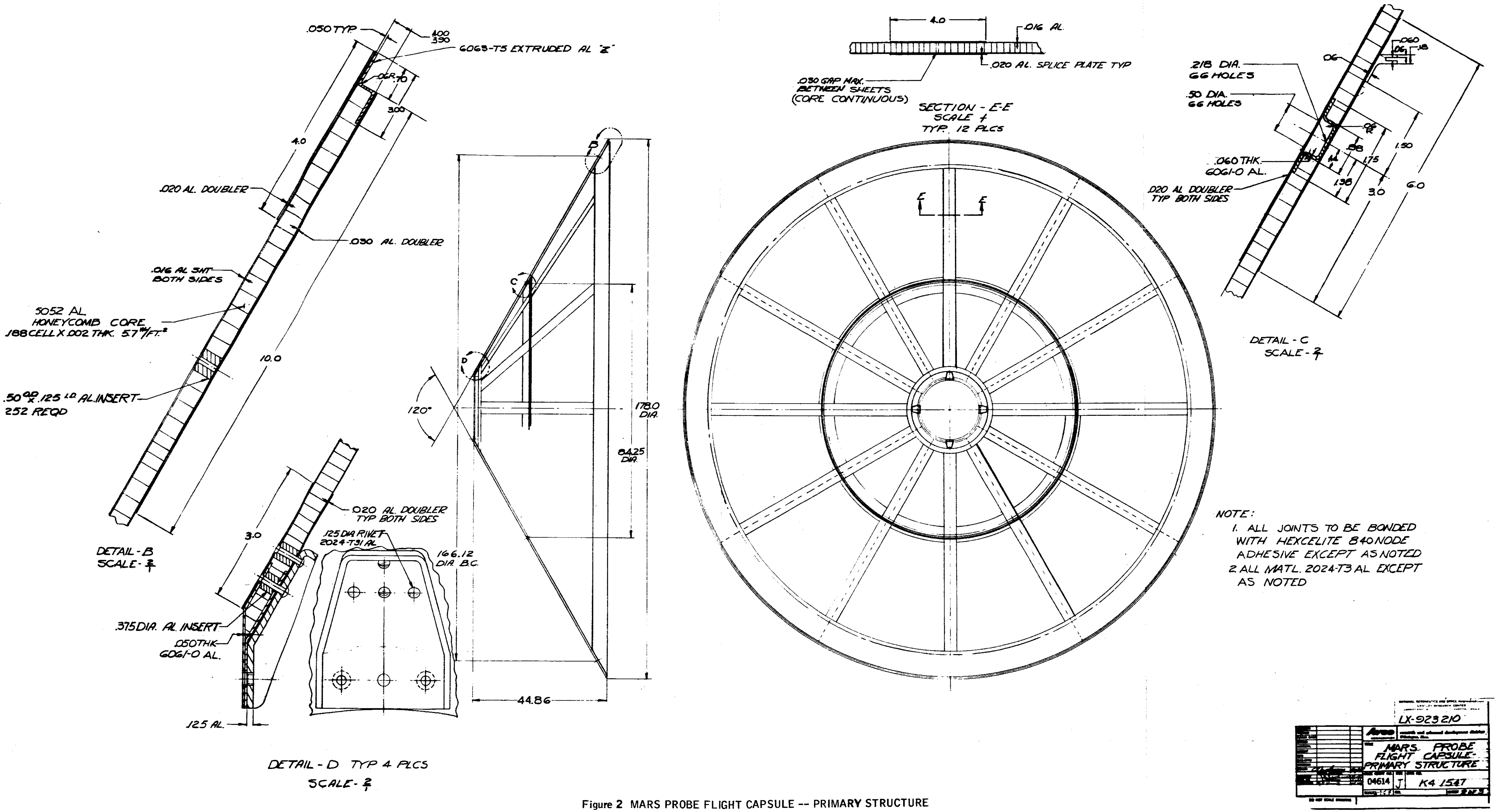


Figure 2 MARS PROBE FLIGHT CAPSULE -- PRIMARY STRUCTURE

259-1

ENTRY SHELL STRUCTURE COMPONENT SPECIFICATION

Number III-2-2-T

1.0 SCOPE

This document specifies the performance and design requirements of the entry shell structure as a component of the Flight Capsule entry shell subsystem.

2.0 APPLICABLE DOCUMENTS

2.1 Flight Capsule Mission and System Specification, Final Report Volume III, Book 2, Part I.

3.0 PERFORMANCE AND DESIGN REQUIREMENTS

The entry shell structural design shown in figures 1 and 2 shall be in accordance with the performance requirements of 2.1.